AFOSR Scientific Report
AFOSR 67-0938

AD 655 792

FLAME-PILOTING MECHANISMS in LIQUID PROPELLANT ROCKET ENGINES

AFOSR SCIENTIFIC REPORT

OCTOBER 1966

Prepared Under Contract AF 49(638)-1120

For

DEPARTMENT OF THE AIR FORCE
AIR FORCE OFFICE OF SCIENTIFIC RESEARCH
WASHINGTON, D. C.

Qualified requestors may obtain additional copies
from the Defense Documentation Center.

AFOSR Scientific Report
AFOSR 67-0938

FLAME-PILOTING MECHANISMS

in

LIQUID PROPELLANT ROCKET ENGINES

October 1966

Prepared Under Contract AF 49(638)-1120

For

Department of the Air Force
Air Force Office of Scientific Research
Washington, D. C.

Aubrey C. Tobey and E. Karl Bastress

Arthur D. Little, Inc.
Cambridge, Massachusetts

Distribution of this Document is Unlimited.

FOREWARD

This program was sponsored by the Air Force Office of Scientific Research and monitored by Dr. Bernard T. Wolfson, Propulsion Division, Directorate of Engineering Sciences.

ABSTRACT

This program was directed toward the systematic investigation of flame piloting mechanisms in liquid rocket engines and their role in combustion instability. The mechanism which gained the most attention was recirculation. The program included experimental analysis of rocket engine performance as affected by external disturbances, minor geometry changes, and propellant state changes: and theoretical analysis of mixing in chemically reacting systems.

Principal conclusions drawn from the results of the program are:

a. Characteristics of the injector and of the propellants are principal factors in the chemical kinetics and fluid dynamics within rocket engine combustors.

b. For the rocket engine and perturbation technique used, engine stability is unaffected by perturbations of the recirculation zone. This does not necessarily eliminate the process of recirculation as a factor in engine stability.

TABLE OF CONTENTS

LIST OF FIGURES

I. SUMMARY

This program was directed toward the systematic investigation of flame piloting mechanisms in liquid rocket engines and their role in combustion instability. The mechanism which gained the most attention was recirculation.

Past work described flame piloting mechanisms only qualitatively. Their influence on engine stability, though apparent, had not been examined systematically. No attempt had been made to relate the work done in gaseous engines with that of liquid engines. The approach in this program was an attempt to isolate as many parameters as possible, vary each in a controlled fashion, and establish their critical nature in total system performance.

The program plan included experimental analysis of rocket engine performance as affected by external disturbances, minor geometry changes, and propellant state changes; and theoretical analysis of the mixing in chemically reacting systems.

Initial work centered around the selection of the basic engine geometry, the propellant combination, and the experimental and data reduction techniques to be used. The engine consisted of a small constant diameter, variable length combustion chamber, with centrally located coaxial injector. The coaxial injector was selected because of its importance in large engine design, and because prior to this work, no comparison had been made of mixing within reacting and non-reacting, confined coaxial streams. This initial geometry was to be changed later to a multiple coaxial injector system to determine the effect of jet interaction as another independent variable.

Methane and oxygen were selected as the initial propellant combination. This was to change to hydrogen and oxygen. Such a procedure would permit the simple transition from the gas state combination through the gas-liquid and finally the liquid combination. These combinations were selected because they appeared to be the easiest to work with in making direct comparisons of mixing and combustion characteristics within geometrically similar systems for which the propellant state was varied. Both hydrogen and methane are of great interest to engine designers. Additionally the substitution of hydrogen for methane would introduce another independent variable into the kinetics of combustion.

In the actual experimental program, tests were conducted to determine concentration, temperature, and chamber pressure variations as functions of mixture ratio, chamber length, injector configuration, and inert gas injection location and rate. A concentric coaxial injector scheme variously modified to accept gaseous methane and gaseous or liquid oxygen was used. High speed motion pictures were taken of the flow dynamics as caused by changes of independent variables. Spectrographic analyses of gas samples withdrawn from various positions along the length and circumference of the engine served to define time averaged concentration gradients of the propellants in the non-reacting case and of the stable products of combustion for the reacting case. Inert gas injection provided the "external disturbance" parameter. Temperature measurements were attempted either by use of the pneumatic probe or the thermocouple probe.

Miscellaneous tests consisted of cold flow free and confined concentric coaxial jet operation with water, liquid nitrogen, liquid oxygen initially saturated or subcooled, with and without gas injection; ignition studies to establish transient conditions resulting from cryogenic fluid state, and particle and droplet tracking experiments to obtain velocity profiles near the injector face.

During the initial phase of the program, a conceptual model consisting of well-stirred and plug flow reactors was proposed to demonstrate the kinetic and flow processes occurring within the system. The proposed model attempted to account for both the chemically and the mixing-limited cases. The actual processes were too complex for valid application of this concept. Experimental data showed the burning rate of a confined coaxial jet to be controlled by diffusion of the reactive species into a time averaged flame front. Therefore, consideration was given to a diffusion model. Satisfactory solution was not readily apparent since velocity profiles, needed for momentum transfer functions, were missing.

The results of this program indicated that:

a. Characteristics of the injector and of the propellants are principal factors in the chemical kinetics and fluid dynamics within rocket engine combustors.

b. For the rocket engine and perturbation technique used, the engine stability was unaffected by perturbations of the recirculation zone. This, by no means, eliminates the process of recirculation as a factor in engine stability.

c. The energy release distribution in the gas-liquid fed chamber, though grossly similar to that for the gas fed rocket is quite dissimilar in detail.

d. At the same mixture ratio, the stability of a gas fed rocket is greater than that of a similar geometry and mass flow rate liquid-gas fed rocket.

e. Cold flow tests using water at ambient temperatures were not indicative of flow, drop breakup, and size distribution patterns existing for cryogenic fluids under similar flow conditions.

f. Greater attention should be paid to the profound effects of the thermodynamic fluid state at injection on engine stability.

g. Interrelations between fluid injection temperature and injection momentum ratio are uncertain.

Recommendations for future study include:

1. A study of multiple coaxial injector systems under conditions exactly similar to those of this study to establish the effect of jet interaction.

2. A study of rocket stability as affected by controlled periodic phase changes of the cryogenic fluid during any given test run. Supercritical fluid properties should be investigated in relation to engine performance.

3. A study of the effect of engine geometry variation on recirculation zone size and sensitivity. Changing length to diameter ratios and engine diameter to injector diameter should be investigated. Then a similar study with multiple injectors should be undertaken. Additionally, electrical analog techniques should be employed for examination of feed back mechanisms and their role in oscillatory systems.

4. A continuation of studies as outlined to provide for a truly systematic investigation of the flame piloting mechanisms and their role in engine stability.

The program has produced some interesting and informative results relating chemical kinetic and mixing processes in a reacting rocket system. Some of its phases should be continued to assist in integrating the work of the small engine research with the needs of working engine designers.

II. INTRODUCTION

The occurrence of uncontrolled high frequency combustion oscillations is of major concern in the development of large high-thrust liquid propellant rocket engines. In order to arrive at rational engineering principles for achieving dependably stable engine designs, it is first necessary to know the mechanisms by which harmful oscillations are excited and sustained, and then to apply this understanding to the development of methods of control.

Although a tremendous wealth of empirical information[1] exists concerning injector design, few correlations exist between mixing processes and instability parameters. Theories have been developed relating the appearance of instability to droplet evaporation rates. Available experimental data describe only qualitatively the influence of mixing (interpreted here as various combinations of the processes of molecular and eddy diffusion, recirculation, atomization and vaporization) on engine performance. However, one cannot question that the time and spatial rates of mixing within an engine, to a great extent, determine the validity and importance of any instability model whether describing a small experimental rocket or a large thrust engine using high density injectors. We believe that fundamental data regarding phenomena occurring near the injector face, such as recirculation, gross mixing, atomization and evaporation, are necessary before rational design parameters can be established.

Past work has demonstrated, in many cases, that combustion instabilities are related to some resonant acoustic mode of the chamber. However, the mere knowledge that the oscillations occur in the acoustic modes leaves open the question of the mechanism by which the oscillations are driven. Concerning this mechanism it is under-

stood in a general way that a time lag exists between the injection of a liquid propellant element into the chamber and the combustion of this element. This time lag is composed of the time required for the physical processes of atomization, mixing and evaporation, and chemical reaction. If any phase of this lag is sensitive to pressure waves, it is understandable that feedback may occur between heat release rate and pressure fluctuations with the result that oscillations are sustained and augmented.[2]

It has been suggested that the responsiveness of time lag to pressure results from the pressure dependency of the chemical reaction rate. However, it can be shown[3] that rocket propellants in the vapor state at rocket chamber pressures have characteristic reaction times in the order of only 10^{-7} seconds, whereas, characteristic times for pressure pulses in rocket combustion are in the order of 10^{-3} seconds. Recent work[4] indicates that the mechanism responsible for combustion processes; and that the rates of chemical reactions in the homogeneous gas phase are high enough to enable the chemical system to respond without phase lag to periodic variations of the conditions in the chamber. Another investigation[5] relates inherent stability to the amount of "pressure sensitive energy" available for a given perturbation and the phase relationship between the replenishment requirement and the time period of perturbation. A summary report[6] of experimental investigation of combustion oscillations in gaseous propellant rocket motors concludes that in unpremixed systems the degree of mixing has so predominant an influence on the appearance of oscillations that the effects of both the system geometry and propellant reactivity are overshadowed. Another investigator[7] contends that the initiating mechanism for all forms of high frequency can be traced to the "massive mechanical vibration" of the injector system as driven by the intense sound field surrounding the engine. In another program[8], a model for evaluating the driving and damping

forces for combustion oscillations has been developed in terms of the motions of liquid droplets with respect to a relatively weak wave resulting from some initial disturbance.

These programs demonstrate the importance of definitive knowledge of the occurrences near the injector face of a rocket combustor.

Recirculation is a phenomenon characteristic of all combustion systems in which either the oxidant and fuel are injected through orifices at the closed end of a tube, or a flame holding baffle is present in a flowing stream. Its critical role in airbreathing combustors has been demonstrated adequately. However its contribution to the flame piloting mechanism in a rocket combustor is little understood. Whether on a micro- or macro-scale, recirculation may critically contribute to all other processes near the injector face.

The intent of our program has been to provide analytical and experimental information concerning the flame-piloting mechanism in rocket combustors and the effect of external disturbances on its characteristics. In particular, we have examined the recirculation zone in a coaxially fed rocket system using inert injection techniques. The primary measurements have included those of local composition and temperature, and rocket chamber pressure variation with time. These variables have been studied as functions of: oxygen to fuel ratio; chamber length, injection configuration; inert gas injection rate; and position of inert gas injection relative to the injector face.

III. DESCRIPTION OF EXPERIMENTAL EQUIPMENT

Experimental work has been conducted using a 100-lb rocket thrust chamber fed with gaseous and liquid propellants through axi-symmetrical unit injectors. Inert gas was injected through a peripheral ring, and tests were conducted with the ring placed at different positions along the chamber axis. The chamber has been equipped with probe ports in order to determine local composition at radial positions along the longitudinal axis. An exploded view of the motor is shown in Figure 1. Note that the motor is made in sections to vary its length from 4- to 16-inches, each section being water-cooled and provided with the necessary penetrations for probing. In addition, a pressure transducer, a sampling probe, the sintered metal ring, and one injector are shown in the foreground. Cross-sectional views of four axi-symmetrical unit injectors used in the experimental work are shown in Figures 2 and 3. The first injector consists of coaxial streams of oxidant and fuel. The second injector consists of an annular stream of oxidant impinging at a 30° angle on the central stream of fuel. The third injector essentially subdivides the fuel stream into seven smaller ones, all located symmetrically within the oxidant stream. The fourth injector was designed for liquid-gas injection. A critical flow orifice and a cavitating venturi are incorporated in the injector to provide flow rate control upstream of the injection point. The injector also is fitted with two parallel .040 inch diameter probes to determine temperatures or to remove gas samples in the injection region. Mass flow rates of gaseous propellants were determined from knowledge of the injector upstream pressure, using calibration curves obtained with standard ASME square-edged orifices.

The primary instrumentation consisted of (Dynisco PT-25) pressure transducers for transient pressure measurements: a (Consolidated Electrodynamics Corporation Model 5-124) recording oscillograph for receiving electrical signals from the transducers and thermocouples; a (Fastax) high-speed motion picture camera; an (Amperex) tape recorder and (General Radio) sound analyzer; a water cooled sampling probe; and (Consolidated Electrodynamics Corporation Model 610) mass spectrometer for analyzing the samples withdrawn from the motor.

IV. EXPERIMENTAL PROGRAM

A. Program Description

The experimental program consisted of firing rocket chambers with or without inert gas injection, collecting and spectrographically analyzing samples taken at positions along the combustor length and radius, and measuring temperatures by means of thermocouple probes in the injection region. In addition, cold and hot flow tests using a transparent cylindrical motor have been made to provide a better visualization of flow patterns during combustion. High speed camera records were made of ignition and steady operation in normal firings.

Initial studies were conducted using gaseous methane and oxygen as propellants. Later in the program, two liquid-gas propellant combinations were used: liquid oxygen and gaseous methane, and liquid propane and gaseous oxygen.

B. Gaseous Propellants

1. Composition Measurements

Experiments with gaseous methane and oxygen were conducted in a straight cylindrical combustion chamber of 13-inches length. The chamber pressure and local concentrations were examined as functions of injection configuration and gaseous helium injection rate. Also the effects of fuel-oxidant combinations and the motor geometry on the motor stability and the character of the recirculation zone were examined. The longitudinal and radial concentration profiles and mixture ratio gradients for a stably operating rocket, as affected by the injector configuration, gaseous helium dilution, and motor length, are shown in Figures 4 through 17.

The variation of mixture ratio within the 13-inch length combustion chamber is illustrated in Figure 4 for which the radial position is used as a parameter. At the center line the oxidant-fuel ratio is essentially zero (fuel rich) with little variation from the injector face to a position about one-half the distance to the exhaust nozzle, at which point the ratio begins finally to approach the metered value at the exit plane. At the chamber wall and near the injector face, the value is high (oxygen rich) and rapidly decreases as the exit plane is approached. However a considerable gradient at the exit plane is apparent. Along lines parallel to the axis at radial positions between the center line and the wall, the curve shapes can be reasonably estimated from examination of the gradients along the center line and the wall.

Figures 5 through 7 illustrate the longitudinal concentration gradients for the various combustion-gas constituents. Again the radial position is used as a parameter. The oxygen concentration

gradient, shown in Figure 5, is maximum at the wall and minimum at the center line. The fuel concentration is maximum at the center line and decreases to a minimum at the wall. Figure 6 shows water vapor at a minimum at the wall and gradually increased to a maximum near the radial position of 0.25-inches from the center line and then decreased from there to the center line. Longitudinal concentration gradients of all constituents at the 0.25-inch radius are shown in Figure 7.

From these plots a general scheme of events can be visualized. The rapid rate of disappearance of the methane and the oxygen and the rapid rate of appearance of water vapor in a region bounded by axially parallel lines through the 0.25-inch and the 0.50-inch radial loci and transverse planes at longitudinal positions of 3- and 5-inches indicates this to be a region of maximum reaction. The presence of large amounts of hydrogen and carbon monoxide along the center line indicates a region in which only partial oxidation occurs because of the extreme lack of oxygen. The presence of large amounts of oxygen and small amounts of carbon dioxide along the wall indicates a region of complete oxidation. That mixing is poor is a generally valid conclusion.

2. Helium Injection

Helium was injected into the motor for two purposes: first, to qualitatively determine the recirculation zone length and second, to drive the motor into instability. By injecting helium peripherally at positions 3-, 4-, 7- and 10-inches from the injector face, recirculation was demonstrated to exist to positions 4-inches downstream of the injector face. (These measurements were substantiated with high speed camera records.) Instability was not triggered by dilution of the recirculation zone with helium even though helium

accounted for 35 to 40 percent by volume of the total gases. However, Figure 11 does illustrate the vitiating effect of helium addition. The relative amounts of water vapor and carbon dioxide are reduced while the relative amounts of hydrogen and carbon monoxide are increased with the addition of helium.

3. Comparison of Mixing in Reacting and Non-Reacting Streams

Comparative data between the reacting and non-reacting gaseous flow systems are shown in Figures 12 and 13. The cold flow data indicate a fairly constant concentration of reactants across the combustor at positions of 5.5-inches, 8.5-inches, and 11.5-inches from the injector face. The data at positions 1 1/4-inches, 3 1/2-inches and 4 1/4-inches depict the rapid mixing occurring between the methane and oxygen. For the reacting systems under the same inlet conditions, considerable amounts of oxygen exist in the region next to the wall and large concentrations of hydrogen, carbon monoxide, and methane appear in regions near the center line forcing the conclusion that mixing is rather incomplete.

4. Recirculation Zone Blocking

If recirculation is an important parameter its elimination should grossly affect the combustion process. Therefore, within the gaseous rocket, effort was made to eliminate the recirculation zone by insertion of a solid material in its place. Each of two hollow plastic cylinders, one with a 15-degree half-angle diverging conical passage and the other with a 10-degree half-angle diverging conical passage, were placed extending from the injector face of the transparent motor. In each instance recirculation continued to be evident and ignition and steady operation were obtained. In the first case,

recirculation was very strong. In the second case, the amount of recirculation was considerably reduced with no appreciable detrimental effects. Under these conditions no helium injection was attempted.

5. Nitrogen Dilution

Tests were attempted using air and oxygen enriched air as the oxidant and gaseous methane as the fuel. For the case in which air was used, ignition was unsuccessful at various flow rates, oxygen to fuel ratios, and ignition source locations. When oxygen enriched air (50/50) was used, ignition was achieved and the motor performance good. In only one instance of four runs was instability evidenced.

C. Liquid-Gas Propellant Combinations

1. Gaseous Methane - Liquid Oxygen Rocket Operation

a. Composition Measurements

The gaseous methane-liquid oxygen combination was operated at two different mixture ratios. The first series of tests were conducted with a fuel-lean mixture ratio and samples obtained were spectroscopically analyzed for constituency. The spectroscopic analyses of gas samples withdrawn at positions of 11.5 inches, 8.5, 5.5, 3.25, 2.25 and 1.25 inches downstream of the injector face are summarized in Figures 14, 15, and 16. These samples were taken at an over-all oxygen to fuel ratio of approximately 4.0. A comparison of the mixture ratio gradient for the gas-liquid fed system at two different over-all mixture ratios vs. the gas-gas fed system is shown in Figure 14. Three sets of data show the mixture ratio gradient for: (a) the gas fed system operating at an over-all ratio of 3.0; (b) the gas-liquid fed

system at a ratio of 5.0; and (c) the gas-liquid fed system at a ratio of about 4.0. The first two sets of data were obtained by using injector #1 which was designed for the gas fed rocket. The third set of data was obtained using an injector of the same type designed to give a mixture ratio of 3.0 with liquid oxygen and gaseous fuel. The similarity between the results for the gas fed rocket at a mixture ratio of 3.0 and the results for the liquid-gas fed rocket at approximately the same ratio is remarkable. The comparison is much more favorable than that of the two gas-liquid sets of data. These results might be attributed either to the difference in over-all mixture ratio, or to the striated flow as caused by the very large droplets issuing from the wide annulus of the gas-gas injector. One can conclude from this figure that the mixture ratio patterns are profoundly influenced by the physical state of the reactants at the time of injection. This statement will be discussed in more detail later in this section.

Figure 15 indicates the longitudinal oxygen concentration profile as a function of radial position in the chamber. The results are not totally unlike that for the gaseous system however there are some additional features which point rather dramatically to the more messy mixing which takes place within the gas liquid fed rocket. While the axial oxygen concentration near the center line remains fairly constant, it changes drastically as the wall is approached. At positions close to the injector face, an eight-fold increase in the oxygen concentration occurs. A gradual decrease in the oxygen concentration occurs as the exit plane is approached because of radial diffusion and reaction with methane. Interestingly, however, within a region from about 2.5 inches to 6 inches we note a large dip in the oxygen concentration. Whether this is real or not could only be determined with further measurements. However examination of the high speed pictures of flow indicate this to be a region of very

turbulent mixing. It is within this region that the jets first strike the wall and recirculatory as well as forward flow occurs. As a consequence of this gross mixing a significant increase in the spatial reaction rate occurs. Further downstream we see either a decrease in the apparent reaction rate or a mixing of more oxygen into the system as a result perhaps of evaporation from the wall plus the inherent jet spread. If we can use the rate of disappearance of the oxygen as a measure of the reaction rate, we can intuitively establish mixing as the rate limiting force.

The longitudinal mixture ratio gradient as a function of the radial position is shown in Figure 16. We note that along the center line the mixture ratio gradually increases with distance from the injector face. This is due to the gross and diffusional mixing of the oxygen with the fuel. As the wall is approached from the center line the mixture ratio patterns are less clear. In the region between longitudinal positions of 2.25 and 5.5 inches major changes in mixture ratio take place and from 5.5 inches on out to 11.5 inches all data converge gradually to the over-all mixture ratio.

Figure 17 is a plot of the radial mixture ratio gradient as a function of longitudinal position. The data show very definite trends with regard to the fluid dynamics of the system. At a longitudinal position of 1.25 inches, the mixture ratio gradually increases from the center line to the wall with an increasing rate of change beyond the 0.5 inch radius. At the 5.5 inch longitudinal position there is a hump in the curve at the 0.5 inch radial position and then at 8.5 inches this hump is moved out to a radial position within 0.10 inch of the wall. At the 11.5 inch longitudinal position the radial mixture ratio gradients are approaching a mean value. Some comments concerning the system stability during the sampling runs will be made later in this report.

b. Saturated and Subcooled Cryogenic Liquid Injection Studies

Photographic studies of the coaxial jet fed centrally with gaseous fuel and annularly with liquid oxidant demonstrated several important features. For a case in which gaseous nitrogen was substituted for the methane and water for the liquid oxygen, considerable difference was noted between the flow patterns with and without the gaseous nitrogen injection. When the jets were confined in a transparent tube the recirculation for the water and gaseous nitrogen systems was very similar to that of both the reacting and non-reacting gaseous fueled rocket. The recirculation zone extended for a distance of from 3 to 5 inches downstream of the injector face, followed by a well-developed flow region to the exit nozzle. For the case in which liquid nitrogen was then substituted for the water, two flow regimes were present. Next to the wall, the recirculation zone appeared to extend much further downstream into a region 8 or 9 inches from the injector face while a fully developed flow was apparent in the central core.

Attempts to obtain comparative hot flow pictures were unsuccessful because the strong pressure surges upon ignition destroyed the transparent motor. Films were taken of the cold flow operation of the system from the moment that the oxidant and fuel valves opened to the steady-state operating condition using either saturated or subcooled liquid oxygen. The upstream pressure was 300 psi gage, and the jet was issuing into atomospheric pressure. Initially the jet was gaseous and then fluctuated between this and a liquid state for about 0.6 - 0.75 seconds before settling into a fairly steady stream of liquid droplets and vapor extending for 12-14 inches downstream of the injector face. Slight fluctuations were evident but the stream was quite stable.

Several separate conditions of flow were examined for the oxygen jet whether saturated or subcooled. Those are the free jet, with and without central gas injection, the confined jet, with and without central gas injection, and the confined jet, with central gas injection and combustion. The free or confined jet, whether saturated or subcooled, goes through flow fluctuations the duration of which ranges from 0.6 seconds to 0.75 seconds. These fluctuations are changes in the oxygen phase from gas to liquid and vice versa.

The total duration of the fluctuations for the confined jet appear to be reduced by the central injection of gas. The stabilizing effect of central gas injection may be attributed either to the increase in ambient pressure within the chamber to about 2 atmospheres or to the additional heat flux to the oxygen stream from the central gas stream. If the first premise were true the oxygen would arrive at the injector face as a subcooled liquid and then flash within the chamber. If the second premise were true the oxygen would remain in a gaseous state within the feed system for a longer period of time and the chilldown process would be more uniform with time such that when liquid oxygen finally did appear it would be on a continuous basis.

At this point it would be helpful to describe the above process on a pressure-enthalpy diagram and relate this description to the actual physical process. The oxygen is transferred into a saturated state from a storage dewar to the facility dewar at atmospheric pressure and remains in this facility dewar at either a saturated or subcooled condition depending upon whether the dewar is vacuum or liquid-nitrogen jacketed. Therefore initial point on the pressure enthalpy diagram may lie either on the saturation line or in the subcooled region at 1 atmosphere. Upon pressurization with gaseous helium the oxygen is compressed isothermally to about 20 atmospheres. This process actually subcools the oxygen further. Upon transfer of a slug of

oxygen from the facility dewar to the combustion chamber the liquid oxygen must flow through a vacuum jacketed transfer line into the uncooled injector and then sustain a large pressure drop on the order of 10 - 20 atmospheres into the combustion chamber. During this process the enthalpy is increased slightly by heat picked up in the transfer lines however it may be considered a constant enthalpy process for that portion to expansion through the injector and isentropic during the expansion through the injector. Once the liquid reaches the exit port of the injector it then sustains a large pressure drop and, removing heat from the surrounding gases, flashes into a two-phase mixture of vapor and droplets within the chamber at ambient chamber pressure. More than likely, if the liquid is initially saturated, the flow through the transfer line and the injector will cause the thermodynamic state of the oxygen to terminate in the wet region under the saturated liquid line. For an initially subcooled liquid oxygen the saturated liquid line will not be reached until the oxygen sustains a large pressure drop in passing from the injector exit port into the combustion chamber. Under conditions of hot flow the combustion chamber would be operating at pressures of about 7 atmospheres and the oxygen would as a consequence leave the injector as a subcooled liquid and flash within the combustion chamber. It is reasonable then to expect that, under hot flow conditions, a steady injection rate with no pulsations would be obtained. High speed color films taken of the combustion process do not support this conclusion. They indicate that the flame fluctuates for a period of about 0.75 seconds before it settles down to a steady state process. The half period of the pulsation in one film was between .13 and .15 seconds and ranged from almost complete extinguishment to a flame-filled tube. Under the conditions of almost complete extinguishment an eddy of blue flame was always present next to the injector face. These results indicate that oxygen in the form of a gas is present in the transfer line prior to the opening of the control valve.

In all of the runs for which saturated oxygen was used, the combustion process did settle out into a quasi stable condition. When the transfer lines were cooled with liquid nitrogen such that the oxygen would arrive at the injector face in a subcooled condition, ignition itself was much more difficult and was generally accomplished explosively. Attempts to take high speed pictures of this process were unsuccessful because the high peak pressures upon ignition destroyed the transparent combustion chamber.

c. Reversed Injection Configuration

An injector was built for the same total mass flow rate and mixture ratio range as the one previously used except that the fuel was fed in the annulus and the oxidant centrally. Several features were noticeable. Cold flow pictures of the initially saturated oxygen flow showed the development of condensation shock diamonds as the oxygen flow was initiated. The fluid went through the same pulsations as previously, however, the jet spread was considerably less than that for the reversed case. The hot flow performance was much rougher, ignition was more difficult, and in order to have smooth ignition the over-all mixture ratio had to be much leaner than that for the centrally fed fuel system.

d. Stability Studies

Systematic stability studies for the liquid oxygen gaseous methane rocket were not possible because of the ignition difficulties encountered. The flow pulsations during the initial 0.75 second have a profound influence on the stability of the reacting system. Examination of pressure traces taken during tests show the initiation of combustion instability within the rocket during the time of these violent fluctuations. The cause of this instability was not apparent, however, it continued for a matter of about 0.60 second beyond the starting transients before being damped out. These are the first

indications in the entire program of what might be called true instability.

The attempts to drive the liquid rocket to instability by helium injection tangentially, peripherally and radially were unsuccessful in the rocket fed at an over-all mixture ratio of about 5 to 1. However, during later tests with a mixture ratio in the order of about 4 to 1, these instabilities were present during the period of radial helium injection at a longitudinal position 1.25 inches.

In some runs injection of helium after the initial instabilities were positively damped out caused the reappearance of instability. Termination of this instability seemed to be independent of the inert gas injection cut-off time. During the sampling runs when the probe, located 1.25 inches downstream of the injector face, was inserted into the chamber a distance of about 0.25 - 0.50 inch, pressure surges occurred within the chamber upon ignition which destroyed the pressure transducers either by overpressure or burnout. Whether a direct relationship existed between the injection of helium and the occurrence of instability, or whether it existed because of the location of the probe, was not determined.

A comparison of the pressure variations under stable operating conditions within the rocket using gaseous-gaseous and liquid-gaseous propellant combinations is shown in Figure 18. Obviously the operation with liquid oxygen causes much more noise in the system.

2. Liquid Propane - Gaseous Oxygen Rocket Operation

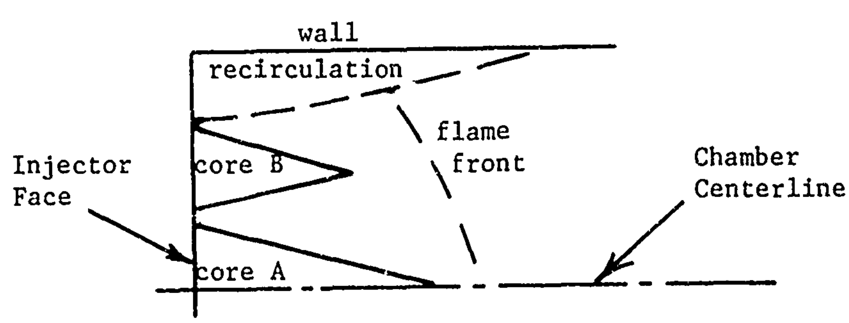
a. Combustion Chamber Performance

Injector #4, shown in Figure 3, was designed for use with liquid propane and gaseous oxygen injected in coaxial jet configuration. Non-cryogenic propellants were selected to avoid transient starting conditions during injector cooling. Operation with this injector and propellant combination was smooth with no ignition problems.

b. Injection Zone Temperature Measurements

This phase of the program consisted of an experimental investigation of the injection region of the rocket combustion chamber. The specific objectives of the effort were: (1) to devise methods for determining the state of the unburned propellants in the injection region, (2) to utilize these methods to determine the dependence of the state of unburned propellants on propellant injection variables, and (3) to relate the state of unburned propellant to combustion stability.

The investigation was based on an intuitive model of the injection region of a coaxial element. The gross features of the model are shown in the sketch below. Features which can be assumed to exist are a central core of propellant A, and an annular core of propellant B. These core regions are composed of undiluted propellant, and the boundary of each can be defined as the locus of points where dilution by external fluids has reached an arbitary value (e.g. 1%).



Another feature which has been shown experimentally to exist is the recirculation region. In this model, the recirculation region can be defined as that volume within which the axial component of fluid motion is in the upstream direction. The boundary of the region is the locus of points at which the (average) axial velocity is zero.

Finally, there is assumed to exist an additional feature described as the flame front. The flame front is defined as the boundary of a region within which combustion reactions proceed rapidly. The boundary is characterized by steep gradients of temperature and composition. The locus of the flame front is unknown, but can be presumed to depend upon combustion chamber geometry and propellant injection characteristics.

The features described above form the boundaries of a region which is described here as the "precombustion zone." Within this zone, various transport processes occur to form a combustible mixture of propellants approaching the flame front. The processes of greatest significance are heat and mass transfer between zones and between fluids of different physical state within each zone. It is postulated that these processes are sensitive to external disturbances, such as pressure waves or injection of inert gases. Such disturbances, by affecting transport processes in the precombustion zone, can cause changes in the rate of energy release in the propellant flame. It is evident that such changes in energy release rate are associated with the various forms of combustion instability which are observed in rocket chambers.

The effort was directed toward an investigation of the precombustion region. The objective was to determine the characteristics of the precombustion region and their dependence upon propellant and combustion chamber variables.

The experimental program was based upon test firings of a rocket combustion chamber with geometric characteristics similar to those of the model described above. The 2-inch diameter chamber used in previous work was employed with injector #4. The injector provides a central fuel orifice with flow controlled upstream by a cavitating venturi. Oxidizer is injected through an annular orifice with flow controlled upstream by a critical orifice.

Basic measurements taken during each test were the following:

1. Oxidizer feed pressure
2. Fuel feed pressure
3. Fuel feed temperature
4. Chamber pressure

These measurements provided indications of propellant flow rates, combustion efficiency, and stability of combustion.

In addition to these basic measurements, temperature probing of the precombustion zone was conducted through the injector as indicated in Figure 3. Thermocouple probes used were .040 inches in diameter with chromel-alumel thermoelements. The probe position was fixed prior to each test, and local temperature was recorded on a high-speed oscillograph.

The test motor was operated with liquid propane (commercial grade) and gaseous oxygen at an oxidizer-fuel mass ratio of 2.54. Motor operating pressure was approximately 90 psia. Propane was injected at $32^{\circ}F$. Since the vapor pressure of propane is approximately 70 psia at this condition, it was assumed to remain as a liquid during injection.

Motor tests were conducted with thermocouples projecting at various distances from the injector face. Test results revealed that

high temperature gases were present very close to the injector face at the thermocouple location. It appeared that an eddy region may have existed downstream of the separating wall between the propellant streams. The thickness of this wall was large in order to accommodate the thermocouples. In later tests, this wall was tapered as shown in the drawing of injector number 4a. It was thought that tapering the wall would eliminate the eddy region and any premature flame-piloting mechanism. However, tests with the modified injector (#4a) indicated that high temperature gases still existed close to the injector. The distance from the injector to the flame front was found to be approximately .15 inches. Since this distance is of the same order as the thermocouple probe diameter (.040"), quantitative measurements of flame front position could not be made with accuracy.

Qualitatively, these tests were significant in that they revealed that a flame-front could exist in a region with no apparent recirculation pattern and unaffected by the recirculation process taking place outside the injector element.

D. Photographic Investigations of Flow Patterns

From the outset of this program the need for good visual characterization of the mixing patterns for free and confined coaxial jets was apparent. Cold and hot flow tests using a transparent motor and a high speed camera were made to provide better visualization of the gross flow patterns during stable operation. For the cold flow free jet investigation, no recirculation was apparent and the air flow patterns about the jet took the classical form of induction patterns. For the investigation of the confined jets recirculation was very prominent. These situations existed whether the system was fed with gaseous methane and oxygen or gaseous methane and liquid oxygen.

These results for the gaseous rocket, shown in Figures 19 and 20, indicated the recirculation zone extent (3- to 5-inches), the region of impingement on the chamber walls, and the fully developed flow field downstream of the impingement areas, thus substantiating the flow patterns estimated from helium injection at various positions along the chamber length.

The photographic studies of gaseous fuel centrally injected and surrounded by an annulus of liquid oxidant demonstrated several important features. For the case in which gaseous nitrogen was substituted for the methane and water for liquid oxygen, the flow patterns with and without the gaseous nitrogen injection were completely different for the coaxial system as shown in Figures 21 and 22. Without the central stream of gaseous nitrogen the annular stream of water extended downstream a distance of between 80 and 100 jet diameters. Only after the addition of gaseous nitrogen did the annular stream diverge and break up into fine droplets, the size of the droplets being a function of the water upstream pressure. When the jets were confined in a transparent tube the recirculation for the water-gaseous nitrogen system was very similar to that of both the reacting and non-reacting gaseous fueled rocket. The recirculation zone extended for a distance of from 3 to 5-inches downstream of the injector face followed by a well developed flow region to the exit nozzle.

Liquid-nitrogen was then substituted for the water. For the free jet condition without central gas injection the liquid nitrogen flashed into a spray as shown in Figure 23. Upon the addition of gaseous nitrogen centrally, the liquid stream dispersion was reduced as shown in Figure 24. This was probably the result of two effects, one due to the aspiration effect of the inner stream on the outside stream, and the second due to the reduced pressure created by the

inner stream condensation at the interface of the two streams. The recirculation zone appeared to extend much further downstream, in fact down to a region 8 or 9 inches from the injector face.

Note should be made of the fact that in all of these experiments the oxidant is injected in the annulus and the fuel into the central tube. At the beginning of this program the work was planned to eventually include a study of liquid oxygen and liquid hydrogen. All work up to that time (1962) appeared to be based on the hydrogen being fed in the annulus and the oxygen in the central stream. Since the diffusion and evaporation rates of hydrogen are much faster than those of oxygen, it seemed reasonable to feed the hydrogen in centrally. We believed this arrangement would flash and disperse the oxygen more readily than the oxygen would the hydrogen. Recent work[9] corroborates our thinking with its conclusions that "with coaxial injection and all operating conditions equal", performance is higher when the less volatile propellant is injected from an annular area surrounding the more volatile propellant. Interestingly, however, the photographic evidence presented in this reference demonstrates the dispersion of the oxygen jet by the central stream of gaseous nitrogen to be somewhere between that of our water and liquid nitrogen results.

This apparent difference depends to a large degree upon the oxygen upstream pressure and the injector design. For example, at this laboratory the liquid nitrogen injection pressures were 300 psia and upon expansion to atmospheric pressure flashing could be expected. For the referenced work oxygen injection pressures were not reported.

V. THEORETICAL STUDIES

A. Conceptual Model

A conceptual model of the occurrences within a rocket chamber will be of some aid in analyzing and interpreting the experimental data. This model must take into account both the mixing and the chemically limited processes. A conceptual model is proposed consisting of a well stirred and plug flow reactors to represent the dynamic and kinetic pressures occurring within the combustion chamber. The purpose of the model is to provide a means of predicting the effect of a recirculation zone vitiation on the general character of the zone as well as its effect on the overall combustor performance.

Mixing rate is controlled by purely mechanical and physical processes occurring in the shear regions between the mixing fluids while the chemical rate is a function of the reactant concentration and temperature. In mixing limited systems the chemical rate is greater by orders of magnitude. Dominance of the overall combustion processes by the chemical reaction rate necessitates prior and complete mixing.

The closest approach to a chemically controlled system occurs in the well-stirred reactor.[10,11,12,13,14] By definition it is a chamber in which composition at the inlet is changed instantly to that of the outlet, and can be characterized by extremely high volumetric heat release rates. Physically, a number of premixed gaseous sonic jets of fuel and oxidant feed into a chamber where fresh reactants are rapidly mixed with burned gases by recirculation and intense turbulence. A simple example[14] of a well-stirred reactor (taken from the reference) is a single step combustion process of the "n^{th}" reaction order involving fuel and oxygen,

$$F + aO_2 \longrightarrow \text{Products} \quad (1)$$

The rate of disappearance of the fuel is represented by the Arrhenius expression

$$- \frac{dN_F}{dt} = Vk\sqrt{T} \quad e^{-E/RT} \quad C_{O2}^{\,n-\alpha} C_F^{\,\alpha} \tag{2}$$

By definition, the temperature within a well-stirred reactor is a function of the exit composition and the inlet temperatures and may be used as a measure of combustion progress or burnedness, β . For a simple one step reaction the final temperature and burnedness are uniquely related.

$$\beta = \frac{T - T_o}{T_{adia} - T_o} \tag{3}$$

Use of this relationship and the equation of state in the Arrhenius expression reduced it to

$$\frac{dN_F}{dt} \quad \frac{1}{P^n V} = \Psi \ (\beta) \tag{4}$$

If the reactor is fed at the molal rate G, with burnedness changing from an inlet value β_o to an outlet value β , the fuel consumption rate is given by

$$\frac{dN_F}{dt} = G \quad (\beta - \beta_o) \tag{5}$$

thus

$$\frac{G}{VP^n} = \frac{\Psi_1 (\beta)}{\beta - \beta_o} \tag{6}$$

the equation for a well-stirred reactor as illustrated in Figure 25.

It can be shown that because of the dilution effects of burned gases on the reaction process, two well-stirred reactors require less total volume than one reactor to accomplish the same reaction. Therefore, the plug-flow reactor concept was developed,[14] and by definition is an infinite series of infinitesimal well-stirred reactors. Its equation then is the integral of

$$\frac{P^n}{G} \quad dV \quad = \quad \frac{d\,\beta}{\psi_1(\beta)} \qquad (7)$$

or

$$\frac{P_\circ^n}{G} \quad (V-V_o) \quad = \quad \int_{\beta_o}^{\beta} \frac{d\,(\ln \beta)}{\psi_2(\beta)} \qquad (8)$$

where

$$\psi_2(\beta) = \frac{\psi_1(\beta)}{\beta} \qquad (9)$$

The plug-flow reactor equation is illustrated in Figure 26.

Only with great difficulty has the phenomenon of recirculation found its way into analytical expressions describing combustion processes. Spalding[15] introduced recirculation as an additional parameter to the conceptual models of combustion systems. Bonnell[16] did an extensive study of the effect of recirculation on the output, and volumetric requirements for well-stirred reactor systems. His method of attack will be used here to gain more insight regarding combustion in rocket motors.

The approach towards a well-stirred reactor and plug-flow reactor by a rocket combustor is influenced by the portion of the chamber devoted to recirculation, and the level of turbulence in the chamber. Clearly, if only a small portion of the chamber is devoted to gross recirculation the combustion rate can be kinetically controlled only through extreme turbulence within the chamber. Even though orders of magnitude difference exist between the reported reaction rates in the well-stirred reactor and those of the rocket combustor, we believe the assumption that the well-stirred reactor can serve as a useful analog is well justified.

B. Application of Well-Stirred Reactor Theory to a Rocket Combustor

A rocket chamber can be divided into several zones which can be best visualized by the Bonnell schematic analog shown in Figure 27. This model is fed unpremixed gaseous fuel and oxidant at a molal rate, G. The following sequence of events occurs:

1 - Fresh feed, G_a, at $\beta = 0$ enters the mixing or shear layer, which is considered a well-stirred reactor, and mixes instantly with recycled burned gases G_a (1 + R), at $\beta = \beta_2$.

2 - This mixture burns from $\beta = \beta_o$ to $\beta = \beta_1$ in V_1. Burnedness $\beta_o = \beta_2$ (R/1 + R) by an energy-mass balance.

3 - The products are discharged from V_1 in two directions:

a - A quantity G_a . R is recirculated to V_2, a well-stirred reactor where it is further burned and recirculated to mix with incoming fresh feed to V_1.

b - A quantity G_a at β_1, mixes instantly with the unburned (β = 0) mixture G_b in the region ($V_1 + V_2$), and then enters a well-stirred reactor V_3.

4 - The mixture reacts to a β which will support a plug-flow reactor V_4.

5 - The mixture is then discharged to V_4, burned to exhaust burnedness, and exhausted.

The volume V_3 represents for the practical case the initial zone or propagation just downstream of the pilot zone. This volume represents the flame-filled tube observed in practical rockets.

These events may be described in terms of the well-stirred and plug-flow curves as in Figure 28. Step one is a simple mixing process of two different temperature gas streams. No reaction is assumed to take place during mixing since the process is instantaneous. The reaction takes place changing the mixture from β_o to β_1, as shown. This is a well-stirred reactor and $\Delta\,(VP^n/G)$ represents the volume V_1 necessary for the combustion process at a constant pressure and molal feed rate. If the reactants should start at zero burnedness, then the reaction must proceed from the origin. The slope of the line from the origin represents the reaction rate, $\Delta\beta\,/\,\Delta\,(VP^n/G)$. The mixture could be dumped into a plug-flow reactor (a system of infinitesimally small isolated well-stirred reactors) or continue the burning process in another well-stirred reactor V_3. In Bonnell's work the recirculation volume V_2 is treated as a well-stirred reactor. If recirculation is pertinent to the total combustion process, this is a good assumption from the standpoint of blowout theory, that as residence time approaches chemical time as a limit, blowout occurs. The volume

V_2 is then a well-stirred reactor receiving β_1 mixture and recirculating a β_2 mixture. The volume required for this can be found from the loading group corresponding to this process $V_2 P^n / G_a R$ where R is the ratio of mixture recirculated to the mixture fed. It should be noted that this recirculation allows a much smaller volume to be used to accomplish the same burnedness that can be managed by a single well-stirred reactor (shown as $V_1 \ P^n/G$).

Part of the material from V_1 is mixed with the fresh material flowing outside the shear region. This mixing process is similar to that at the entrance to V_1, and the resulting burnedness is called β_{02}.

This mixture must go into a well-stirred reactor because a plug-flow reactor is, by definition, fed by a well-stirred reactor and the plug-flow minimum operating efficiency is β_{bo} which occurs at blow-out of a well-stirred reactor. So we burn in V_3 from β_{02} to β_3. The mixture is then fed to volume V_4. This may be either a well-stirred reactor or plug-flow reactor. The plug-flow reactor is chosen because V_4 appears to be more nearly a volume of discrete infinitesimal well-stirred reactors than a single well-stirred reactor.

The above exercise indicates that the possibility exists of predicting the effect of vitiation of the recirculation zone on the general character of the zone, as well as its effect on the overall combustor performance.

C. Jet-Diffusion Model

Experimental results indicate that the burning rate of a confined coaxial jet is controlled by diffusion of the reactive species to a time averaged flame front and that recirculation may be characterized

by a well-stirred reactor. The well-stirred plug-flow reactor theory, however, does not seem to suitably describe the processes occurring within the experimental system of this work. Therefore consideration has been given to a diffusion model of the confined coaxial jet rocket in belief that the combustion instability studies could be considerably aided by a detailed knowledge of the diffusion processes which occur in the immediate vicinity of the injector face. It is here that perturbations are likely to most effectively excite instabilities.

To outline the problem involved in such an analysis consider the simplest case of a single free jet. Furthermore let the jet fluid and the ambient fluid be of the same composition and at essentially the same temperature and pressure. Under such conditions experiments demonstrate the qualitative behavior as follows. At a position close to the injector face, a potential core exists and the jet velocity profile is very flat and cuts off sharply at the edges of the jet. As the fluid proceeds in an axial direction the jet spreads and the velocity decays because of the entrainment of the surrounding fluid and radial diffusion.

At a position far downstream of the injector face, axial stresses are negligible compared to the transverse stresses and the turbulent exchange as occurs by the diffusion process may be empirically modeled. The equation of motion then becomes:

$$u \frac{\partial u}{\partial x} + v \frac{\partial u}{\partial y} = -\frac{1}{\rho} \frac{\partial P}{\partial x} + \frac{\partial}{\partial y}\left(\varepsilon \frac{\partial u}{\partial y}\right)$$

where:

x is downstream distance
y is transverse distance
u is axial velocity
v is transverse velocity
P is pressure
ε is eddy viscosity

This equation is sufficient for axi-symmetric jets. For the case we are discussing, the change of pressure with axial distance equals zero. Experimental evidence concerning the behavior of eddy viscosity may be combined with this equation to yield a quite general relationship of the behavior of a single free jet. A similar analysis of a single ducted jet or confined jet has just recently been successfully carried out.

Returning to the single free jet, consider the shear region which surrounds the potential core. Here the analysis is very complex. Strictly speaking the effect of axial stresses is not negligible and the behavior of the eddy viscosity is more complex. Ignoring these difficulties however, the problem is still almost hopeless because the boundary conditions are complicated and extremely important. For example, the shape of the velocity profile within the injector and the shape of the velocity profile associated with the induced flow over the injector face must be specified.

A satisfactory solution is not readily apparent for either the single free jet or the single confined jet. If one considers the confined coaxial jet, the boundary conditions for the near field profiles are even more complicated and a solution seems quite unlikely. The velocity and concentration profiles are closely coupled and a total solution is not possible without both because the diffusion process cannot be uncoupled from the momentum transfer process. Any solutions which might be obtained will almost certainly be numerical rather than analytical and will probably not convey a clear physical concept of the processes occurring without extremely good experimental information.

REFERENCES

1. Fuhs, A.E., "Spray Formation and Breakup, and Spray Combustion", AMF/TD #1199, Technical Note #4, AFOSR-TN 58-414, 5 February 1958.

2. Crocco, L., Grey, J., Harrje, D.T., "A Theory of Liquid Propellant Rocket Combustion Instability and its Experimental Verification", ARS 827-59.

3. Lewis, B. and von Elbe, G., "Combustion, Flames and Explosions of Gases", Third Edition, Academic Press, New York 1961.

4. Crocco, L., Glassman, I., and Webb, M.J., "Combustion Processes in Rocket Motors", 16th AFOSR Contractors' Meeting on Liquid Rocket Combustion Research, June 25-28, 1963, Nantucket Island, Massachusetts.

5. Peoples, R.G., Baker, P.D., and Knowles, L.S., "High-Frequency Combustion Instability", Aerojet-General Corporation, November 1961, Report No. 2126.

6. Zucrow, M.J., Osborn, J.R., Bonnel, J.M., "Summary of Experimental Investigations of Combustion Pressure Oscillations in Gaseous Propellant Rocket Motors", Final Report No. F-63-2, Contract AF 49 (638)-756, June 1963.

7. McCormack, P.D., "Rough Combustion in Liquid Rocket Motors", Administrative Report #1, Grant No. AF EOAR 63-76, June 1963.

8. Gerstein, M., "Wave Propagation and Droplet Motion in Oscillatory Combustion", 16th AFOSR Contractors' Meeting on Liquid Rocket Combustion Research, June 25-28, 1963, Nantucket Island, Massachusetts.

9. Hersch, M., "Effect of Interchanging Propellants on Rocket Combustor Performance with Coaxial Injection", NASA TN D-2169, February 1964.

10. Longwell, J.P., and Weiss, M.A., "High Temperature Reaction Rates in Hydrocarbon Combustion", Industrial and Engineering Chemistry, 47, 1634-1637 (1955).

11. Avery, W.H., and Hart, R. W., "Combustor Performance with Instantaneous Mixing", Industrial and Engineering Chemistry, 45, 1634-1637 (1953).

12. Hottel, H.C., Williams, G.C., and Baker, M.L., Sixth Symposium on Combustion, Yale University, New Haven, Connecticut (1956).

13. Egerton, Saunders, and Spalding, IME-ASME Joint Conference on Combustion, Massachusetts Institute of Technology, Cambridge, Massachusetts, (1958).

14. Hottel, H.C., Williams, G. C., and Bonnell, A. H., "Application of Well-Stirred Reactor Theory to the Prediction of Combustor Performance", Combustion and Flame Quarterly Journal of the Combustion Institute, Vol. 2, No. 1, March 1958.

15. Spalding, D. B., Imperial College, "Theoretical Relationships Between Combustion Intensity and Pressure Drop for One-Stream Combustion Chambers", C.F. 393, Aeronautical Research Council, Great Britain, 10 April 1957.

16. Bonnell, A.H., Sc.D. Thesis, Massachusetts Institute of Technology, Cambridge, Massachusetts (1958).

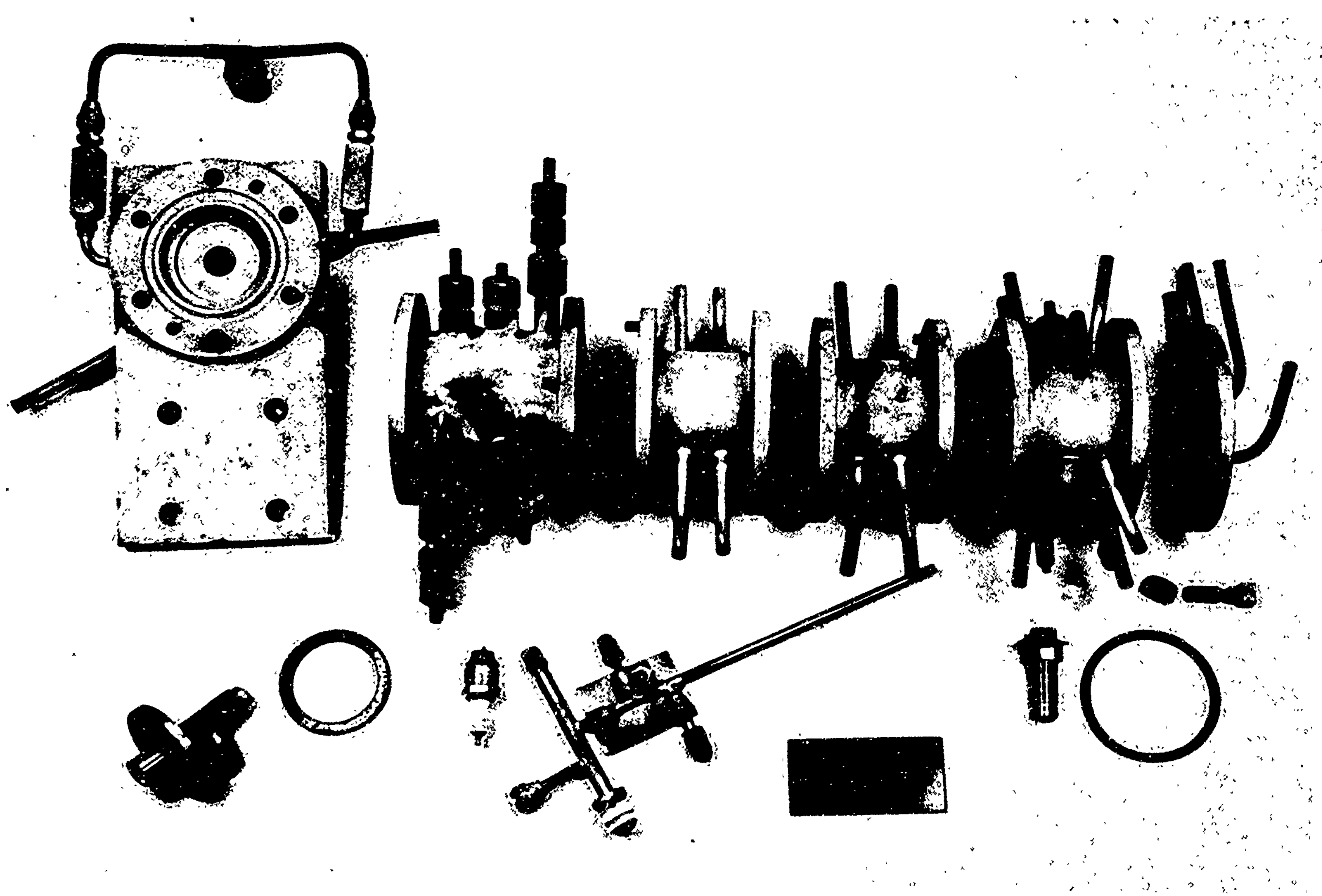
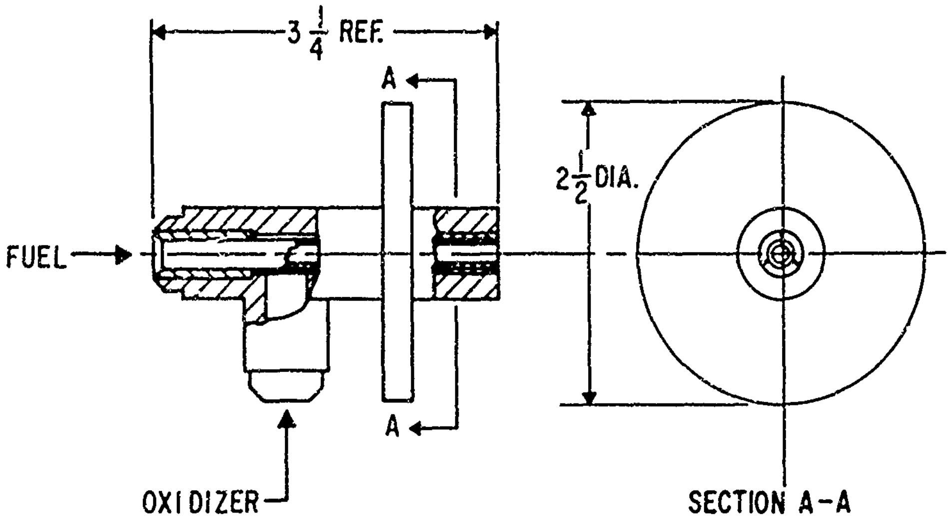
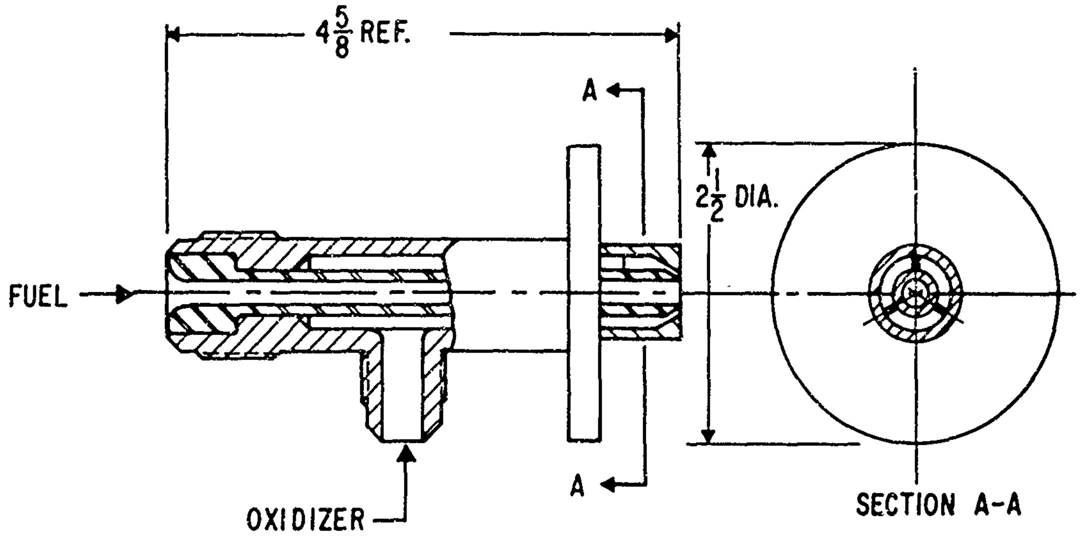


FIGURE 1 EXPLODED VIEW 4 SECTION WATERCOOLED MOTOR-SHOWING EXIT NOZZLE



INJECTOR MODEL # 1



INJECTOR MODEL # 2

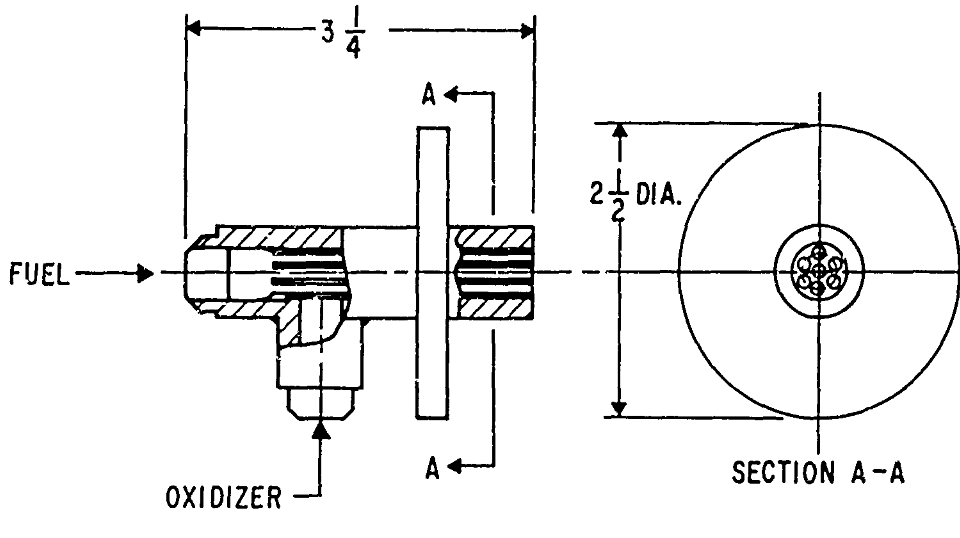


FIGURE 2 INJECTOR MODEL # 3

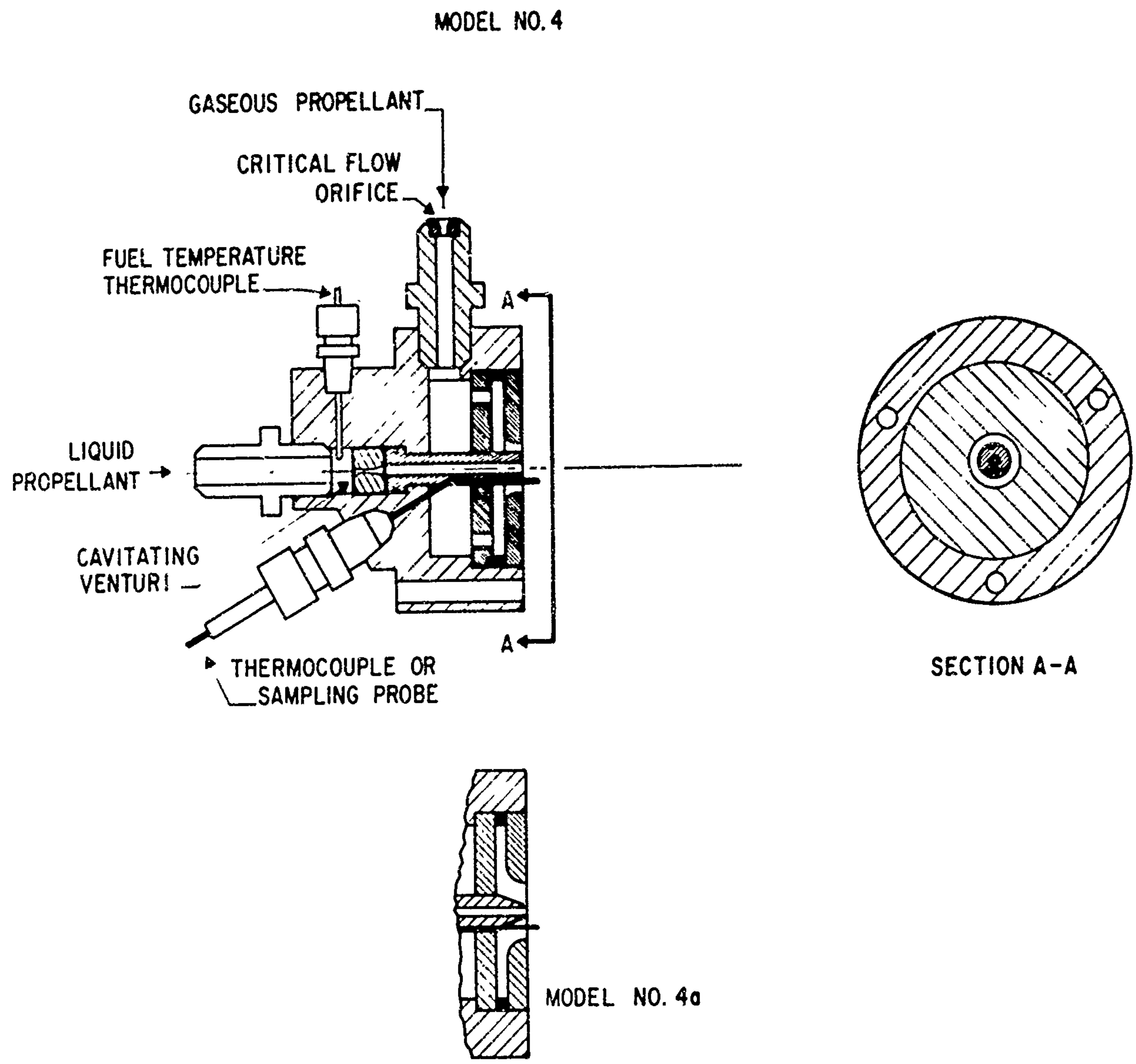


FIGURE 3 INJECTOR MODEL #4

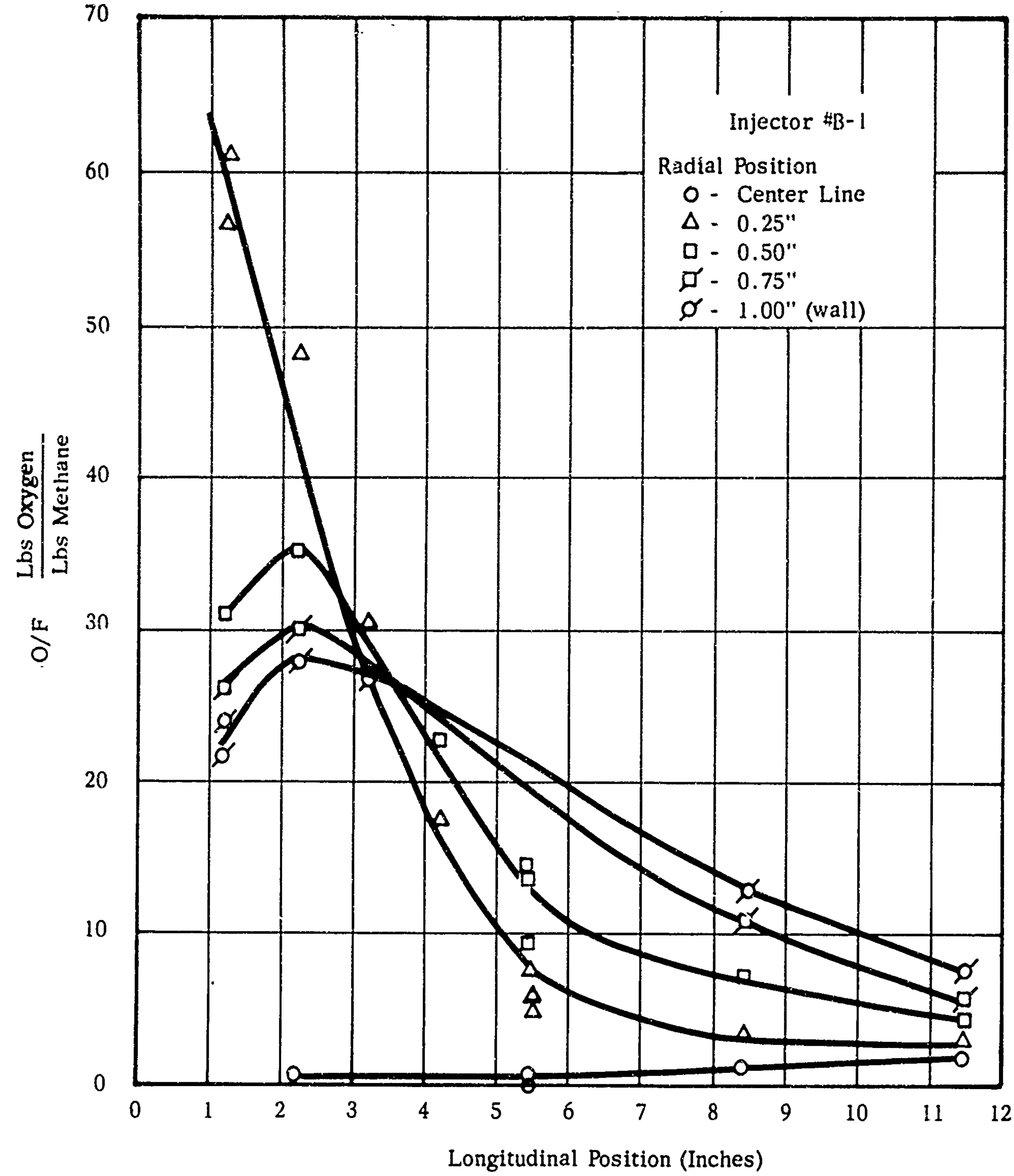


FIGURE 4 13 INCH MOTOR--LONGITUDINAL MIXTURE RATIO GRADIENTS

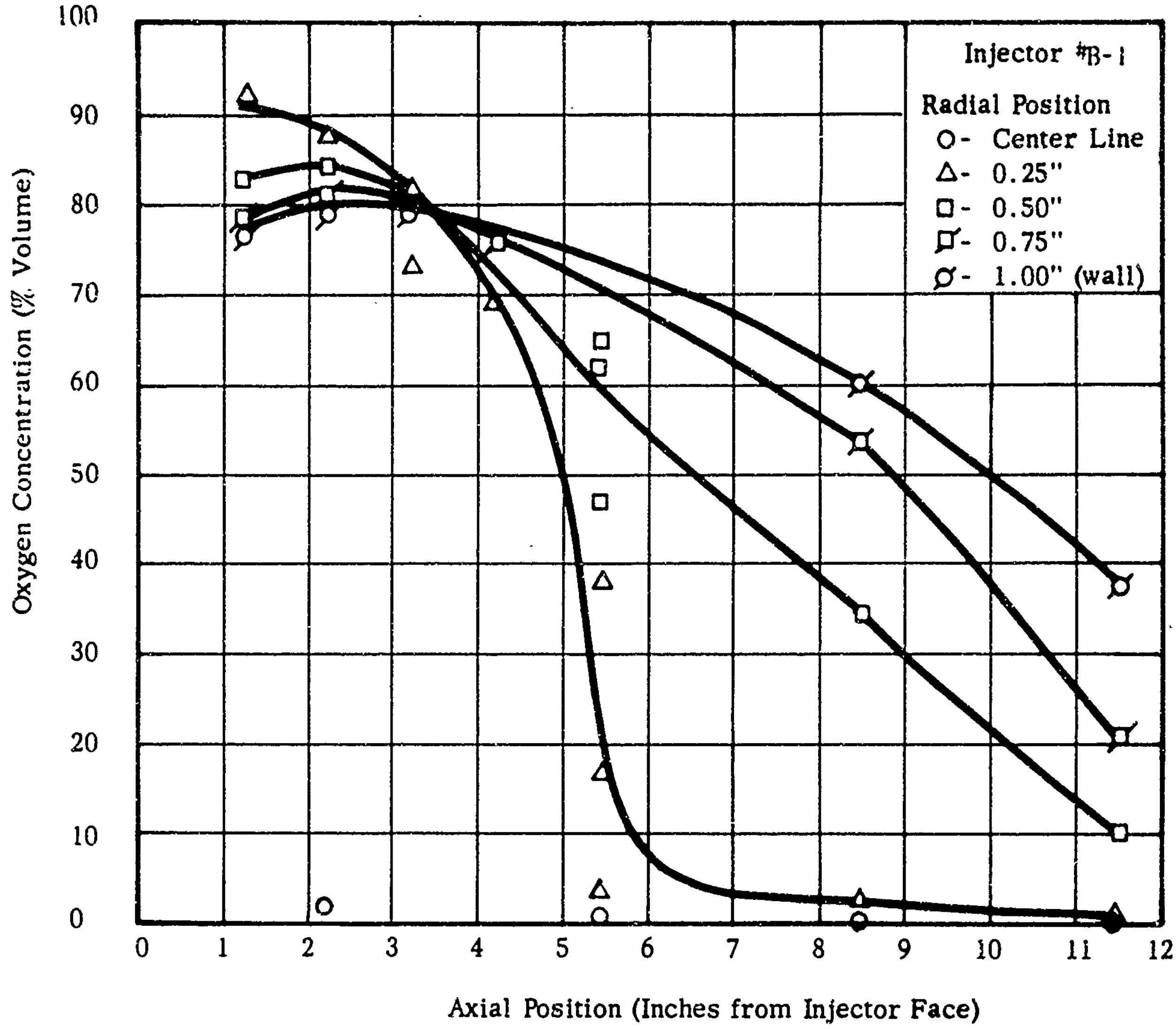


FIGURE 5 13 INCH MOTOR--LONGITUDINAL OXYGEN CONCENTRATION PROFILES

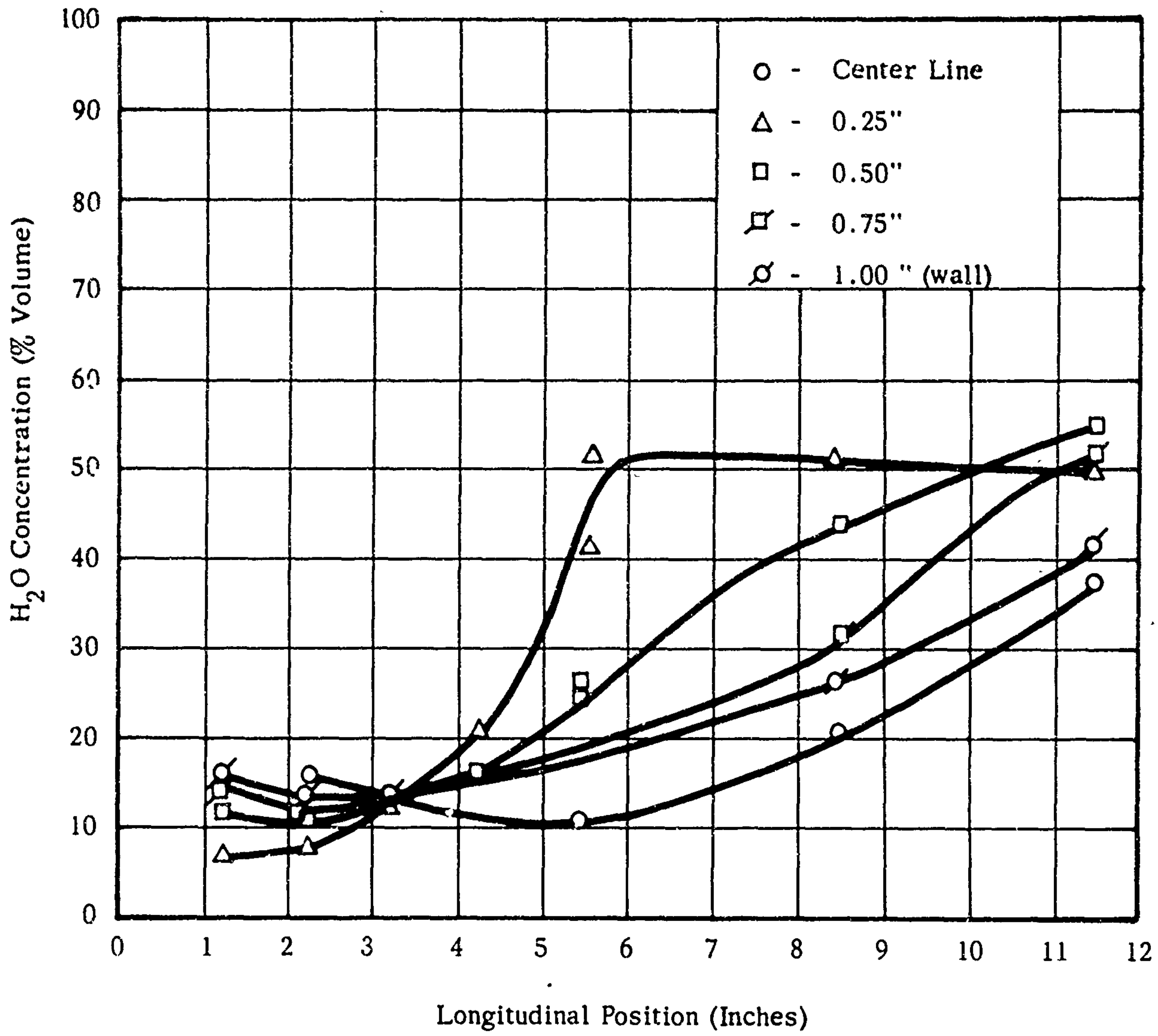


FIGURE 6 13 INCH MOTOR--LONGITUDINAL H_2O CONCENTRATION PROFILES - INJECTOR # B-1

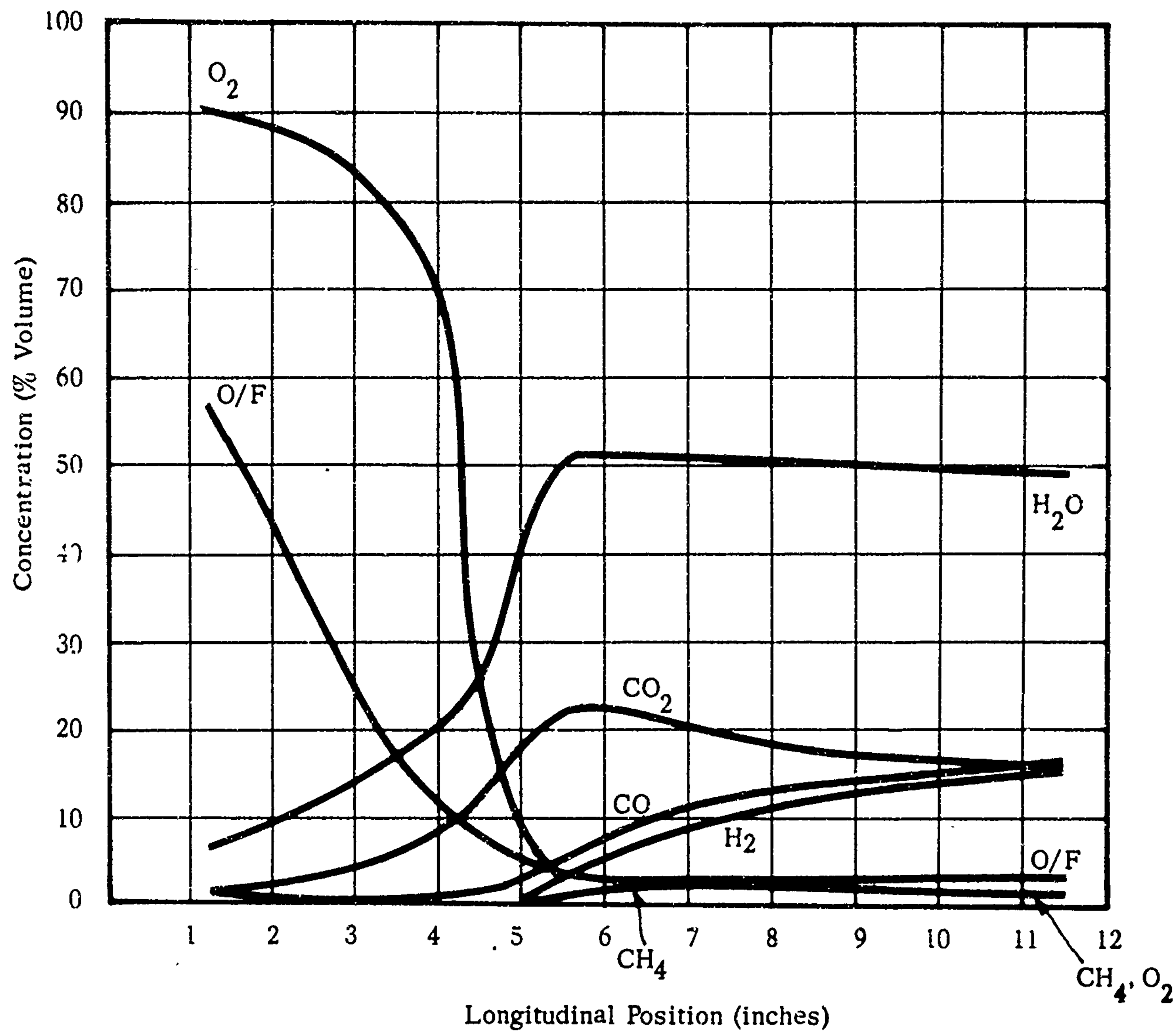


FIGURE 7 13 INCH MOTOR--Longitudinal Concentration Gradients at Radial Position 0.25 inches (Injector #1)

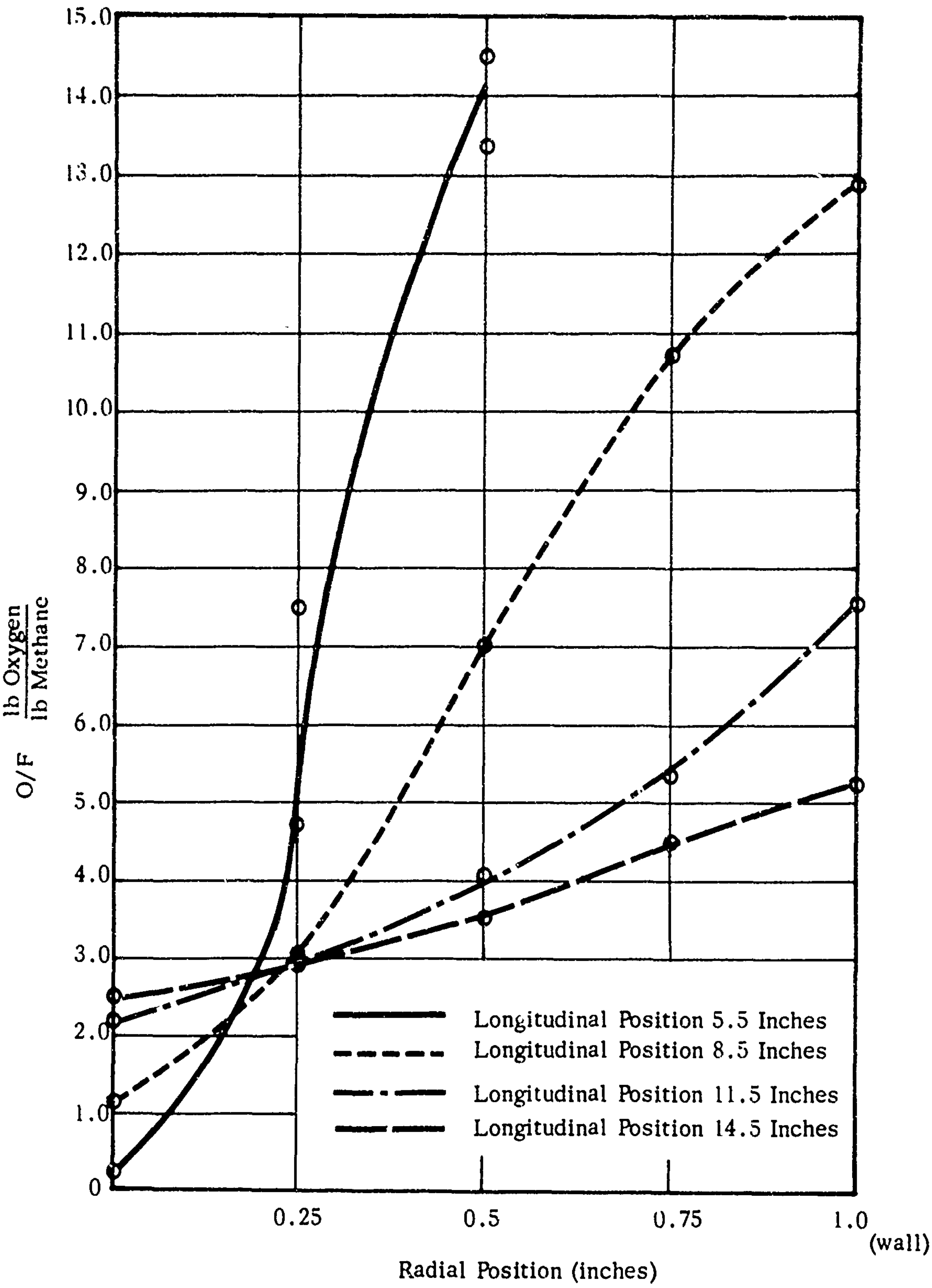


FIGURE 8 RADIAL MIXTURE RATIO GRADIENTS (INJECTOR #1)

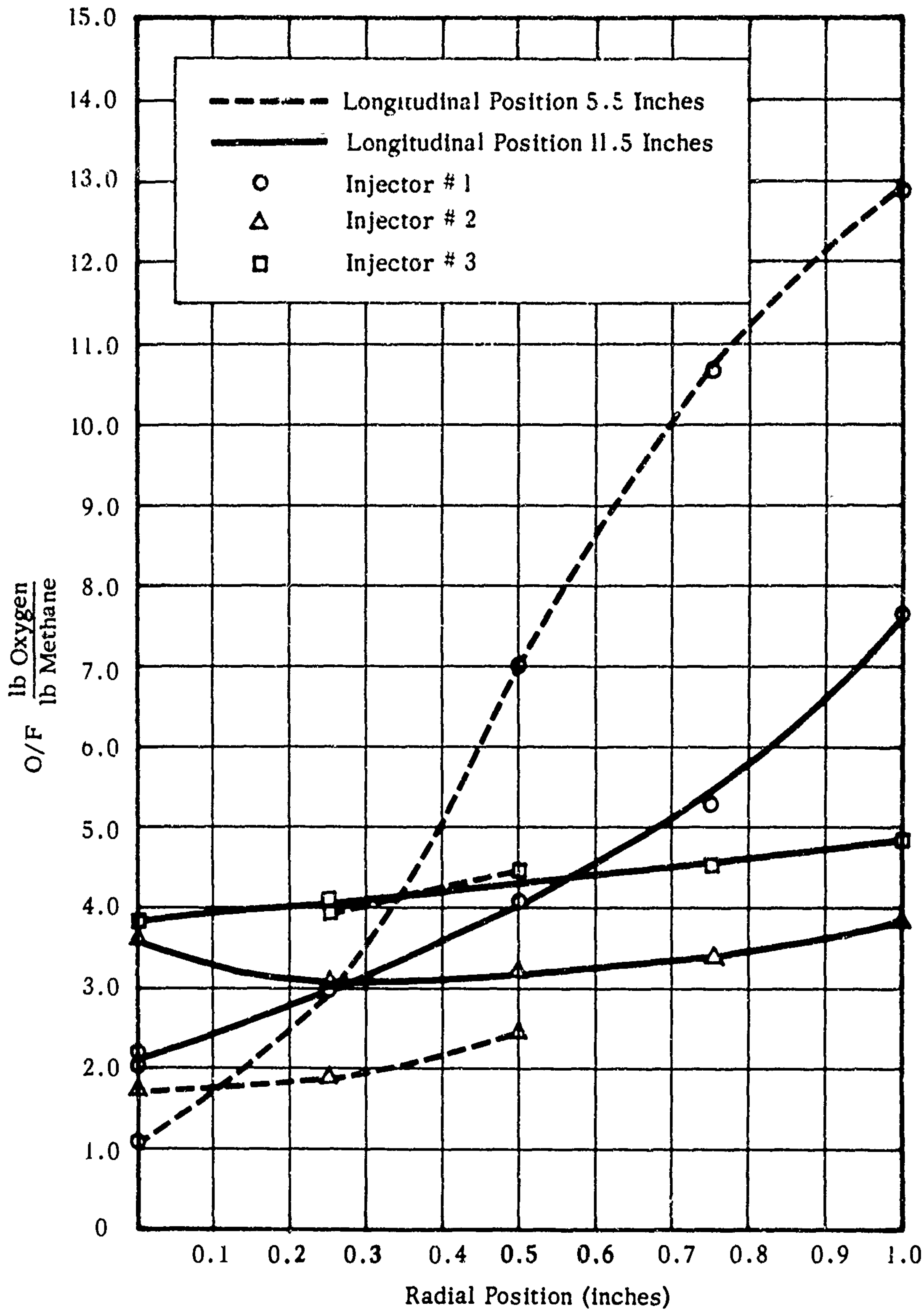


FIGURE 9 RADIAL MIXTURE RATIO GRADIENTS

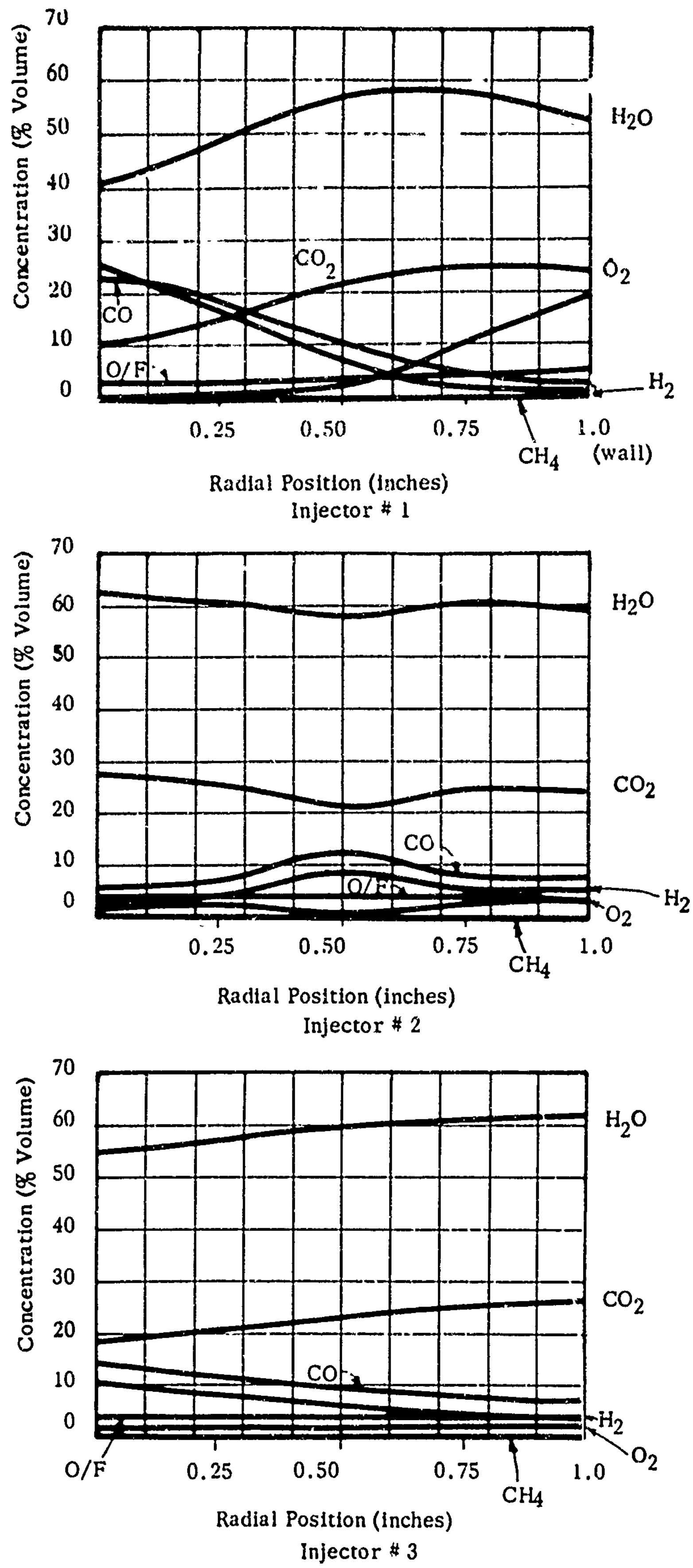


FIGURE 10 MOTOR 16 INCHES -- Radial Concentration Gradients at Longitudinal Position 14.5 Inches

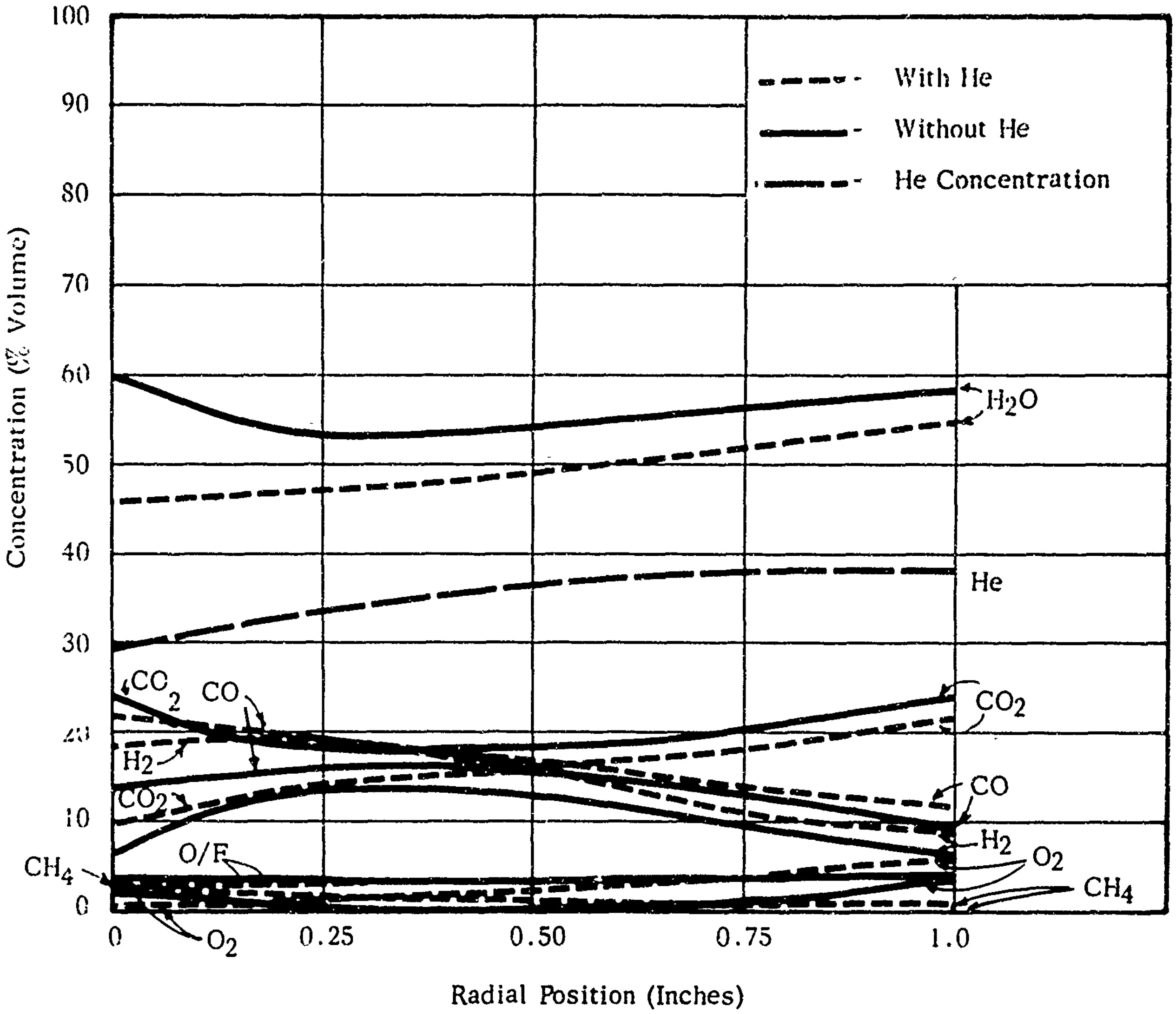


FIGURE 11 MOTOR LENGTH - 13 INCHES--RADIAL CONCENTRATION GRADIENTS @ LONGITUDINAL POSITION 11.5 INCHES (INJECTOR #2)

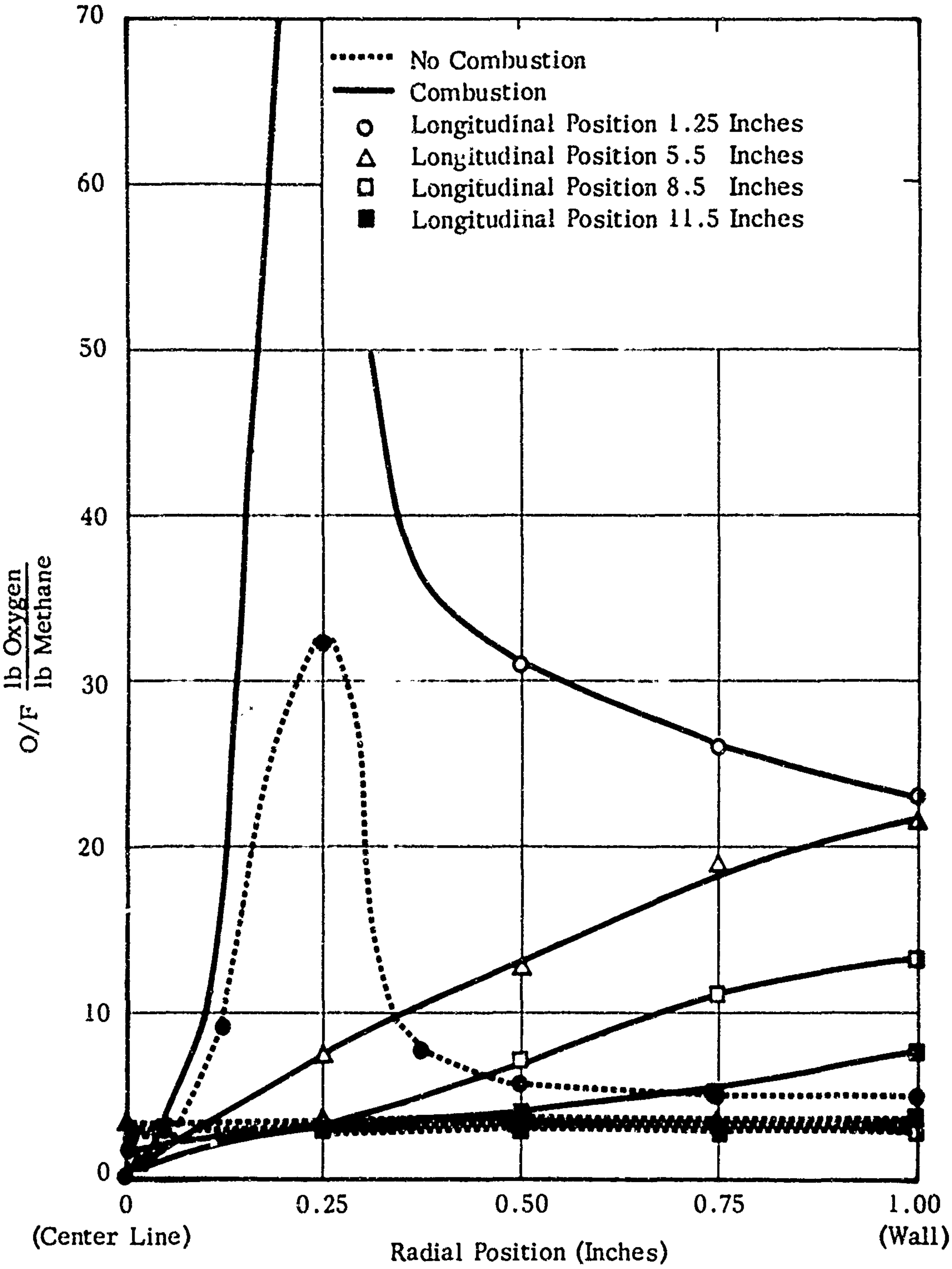


FIGURE 12 13-INCH MOTOR--MIXTURE RATIO GRADIENTS (WITH OR WITHOUT COMBUSTION)

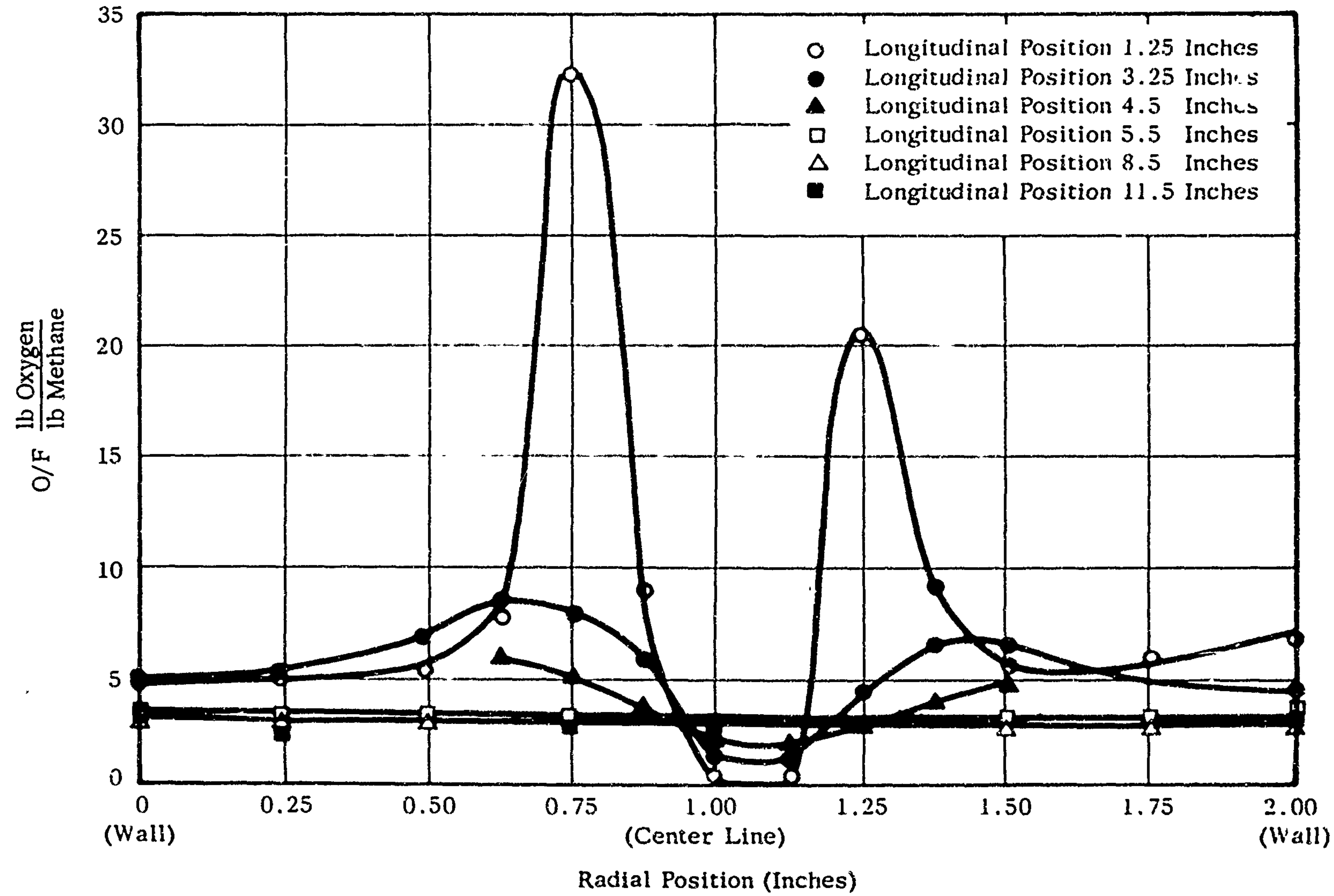


FIGURE 13 13 INCH MOTOR--MIXTURE RATIO GRADIENTS (NO COMBUSTION)

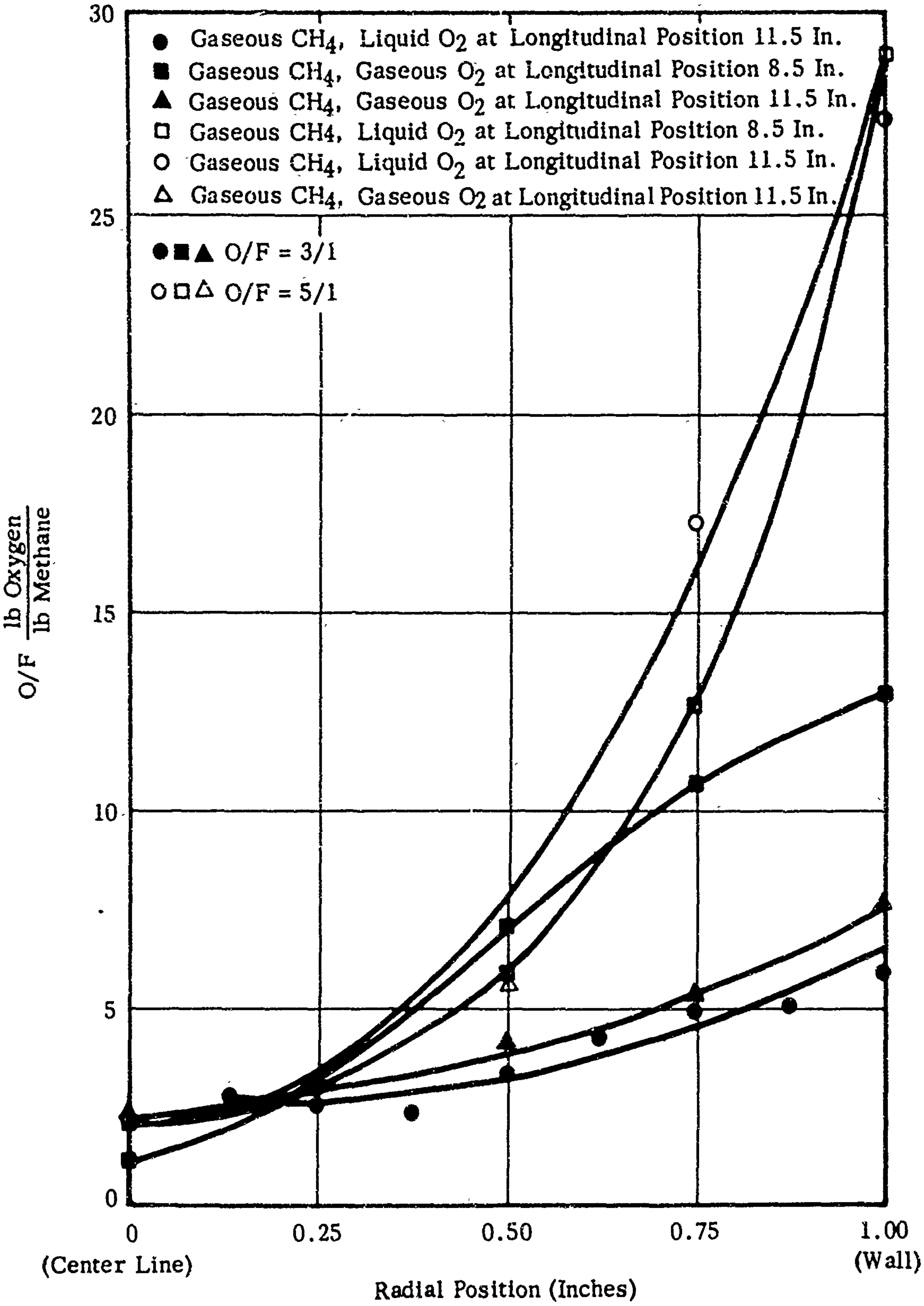


FIGURE 14 13 INCH MOTOR--MIXTURE RATIO GRADIENTS (GASEOUS METHANE AND GASEOUS OR LIQUID OXYGEN)

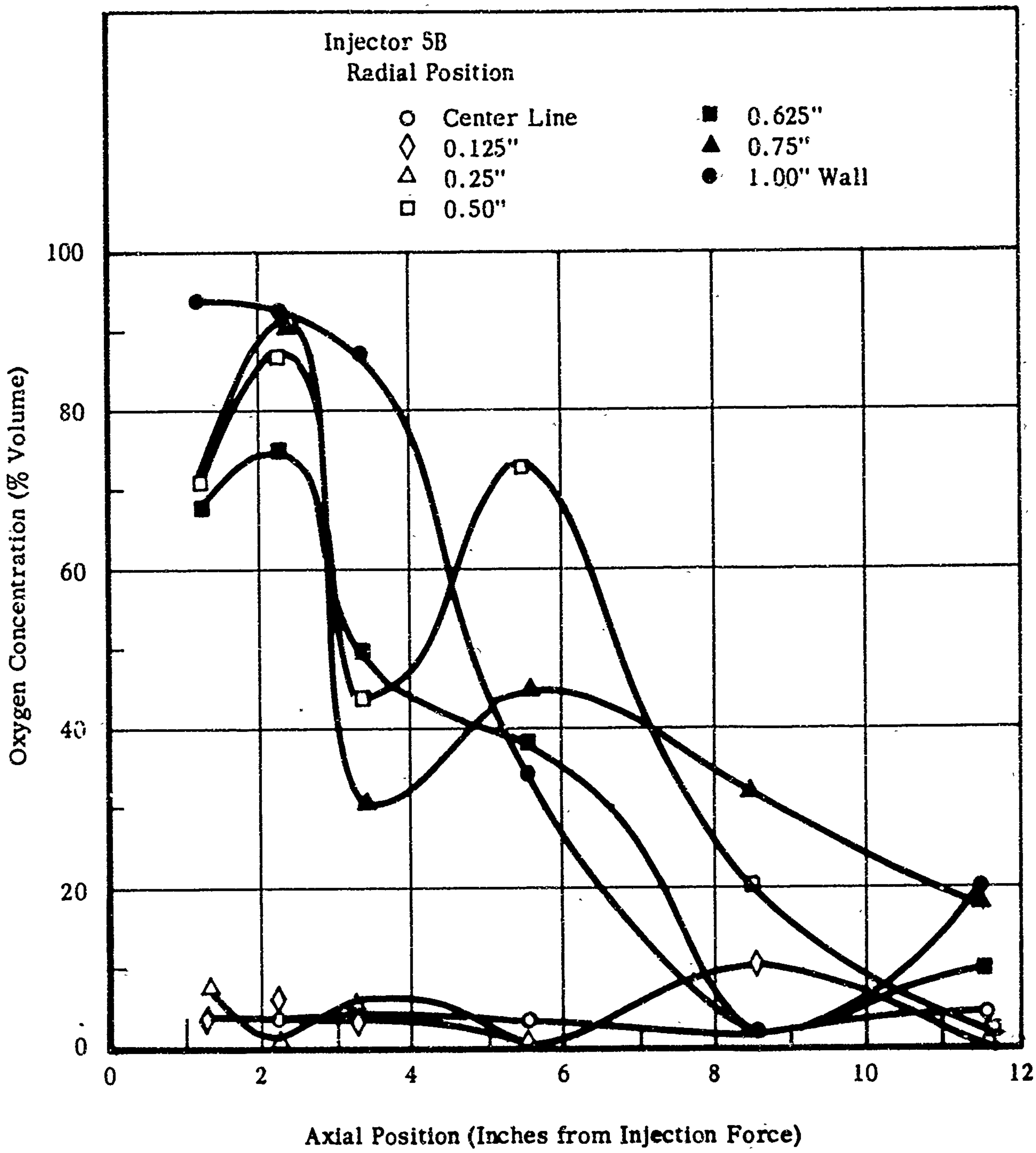


FIGURE 15 13 INCH MOTOR--LONGITUDINAL OXYGEN CONCENTRATION PROFILES
GASEOUS METHANE--LIQUID OXYGEN

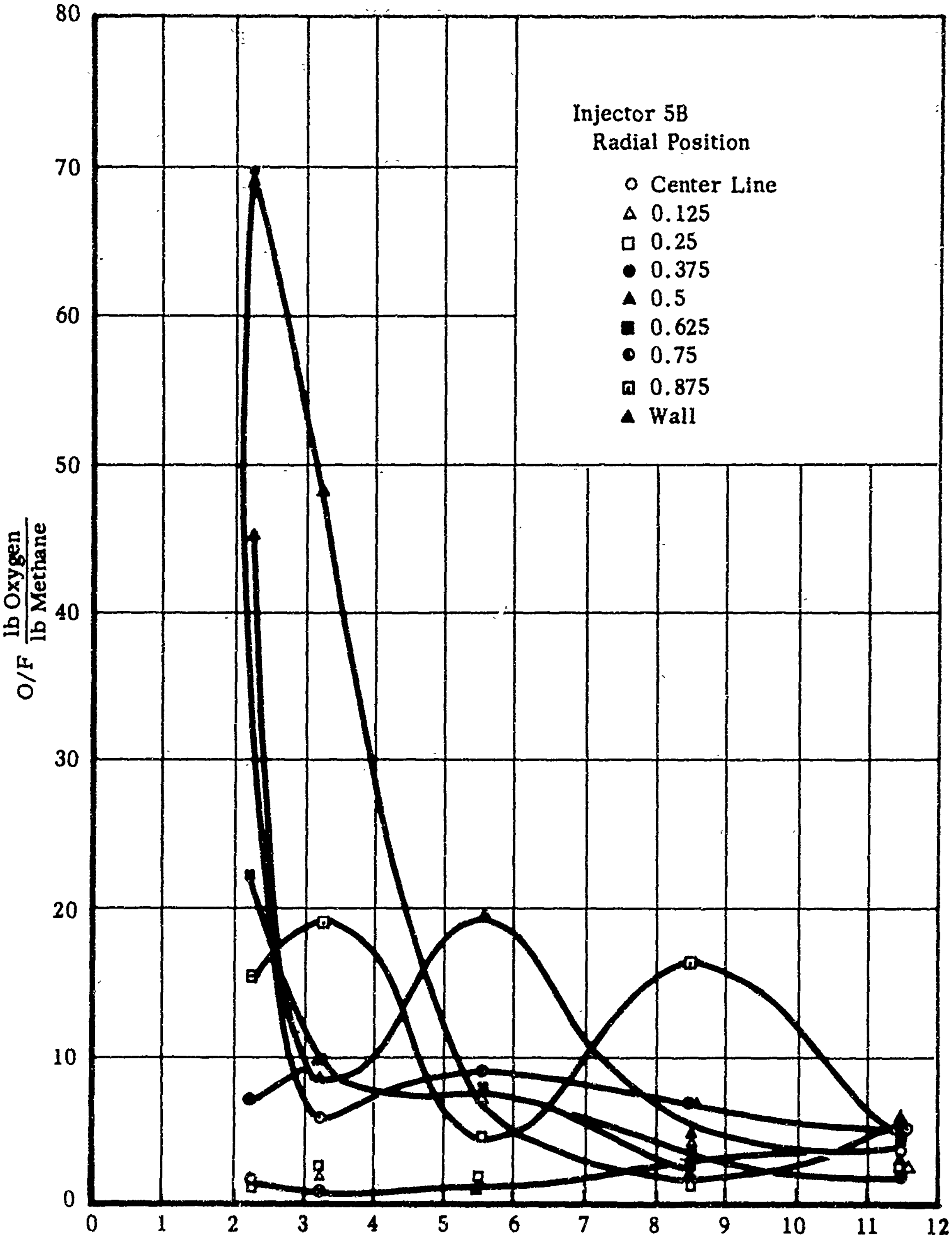


FIGURE 16 13 INCH MOTOR--LONGITUDINAL MIXTURE RATIO GRADIENTS. GASEOUS METHANE--LIQUID OXYGEN

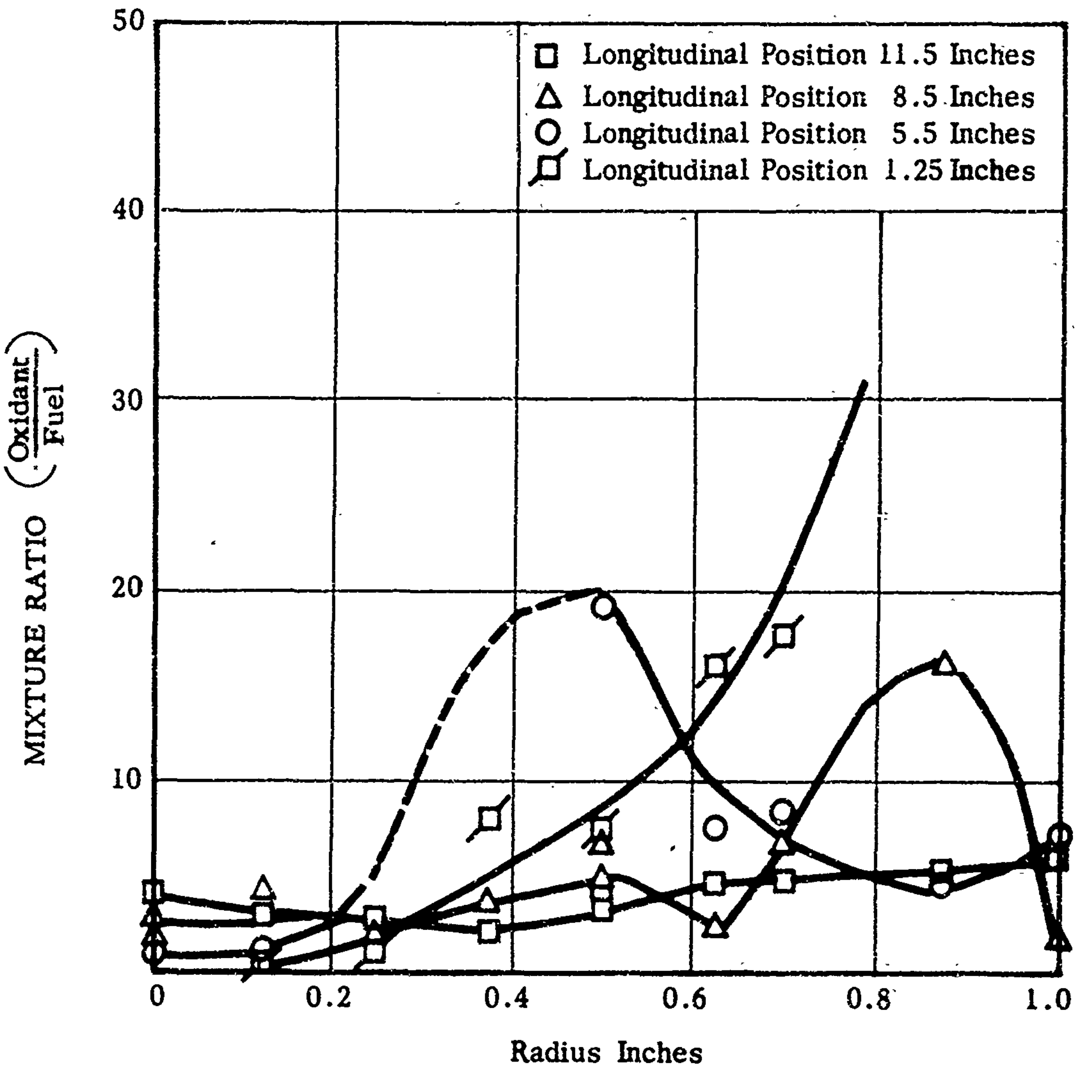


FIGURE 17 RADIAL MIXTURE RATIO GRADIENTS
(O/F) OVERALL 4.0

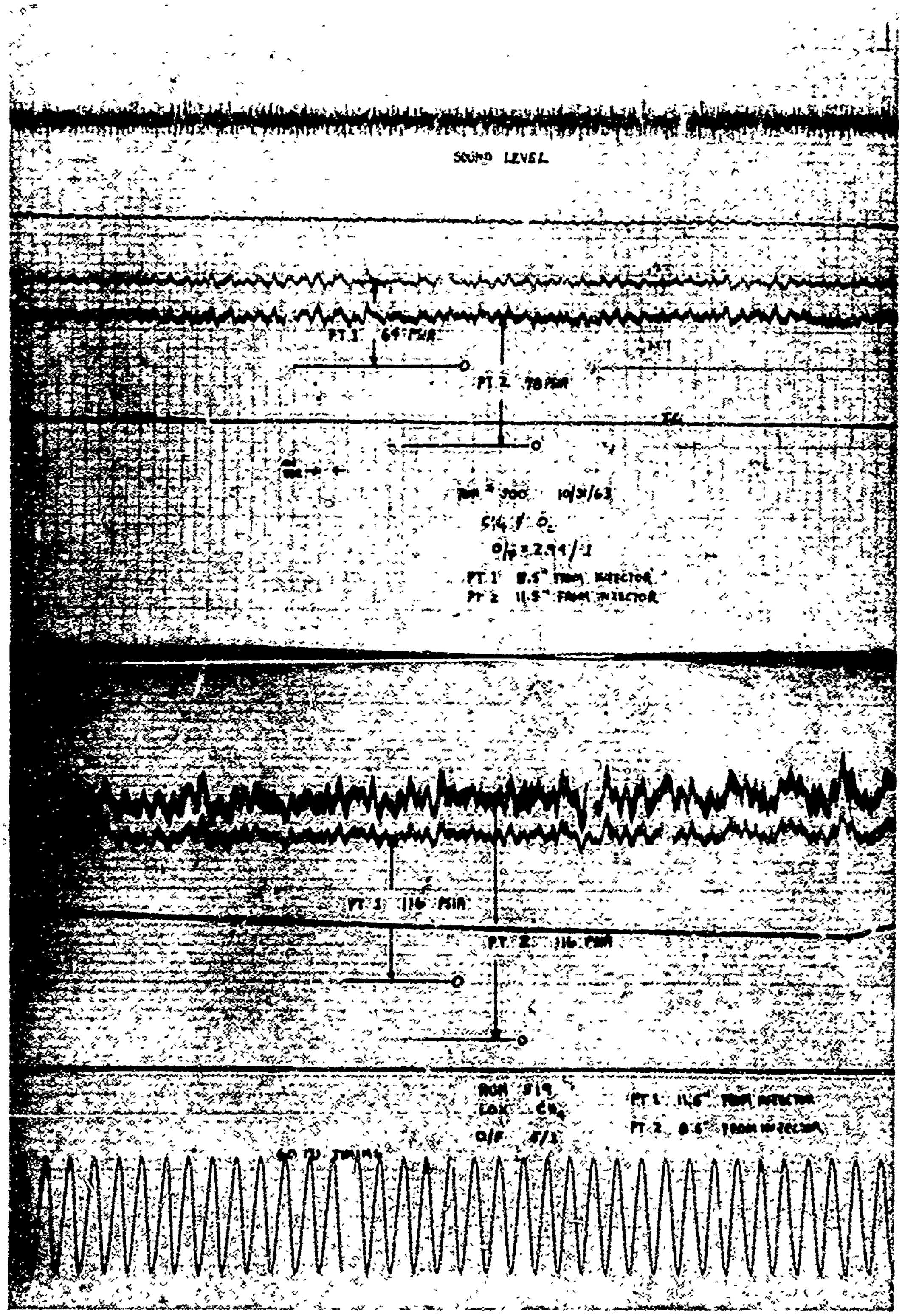


FIGURE 18 COMPARATIVE PRESSURE TRACES OF GASEOUS VS GASEOUS-LIQUID ROCKET

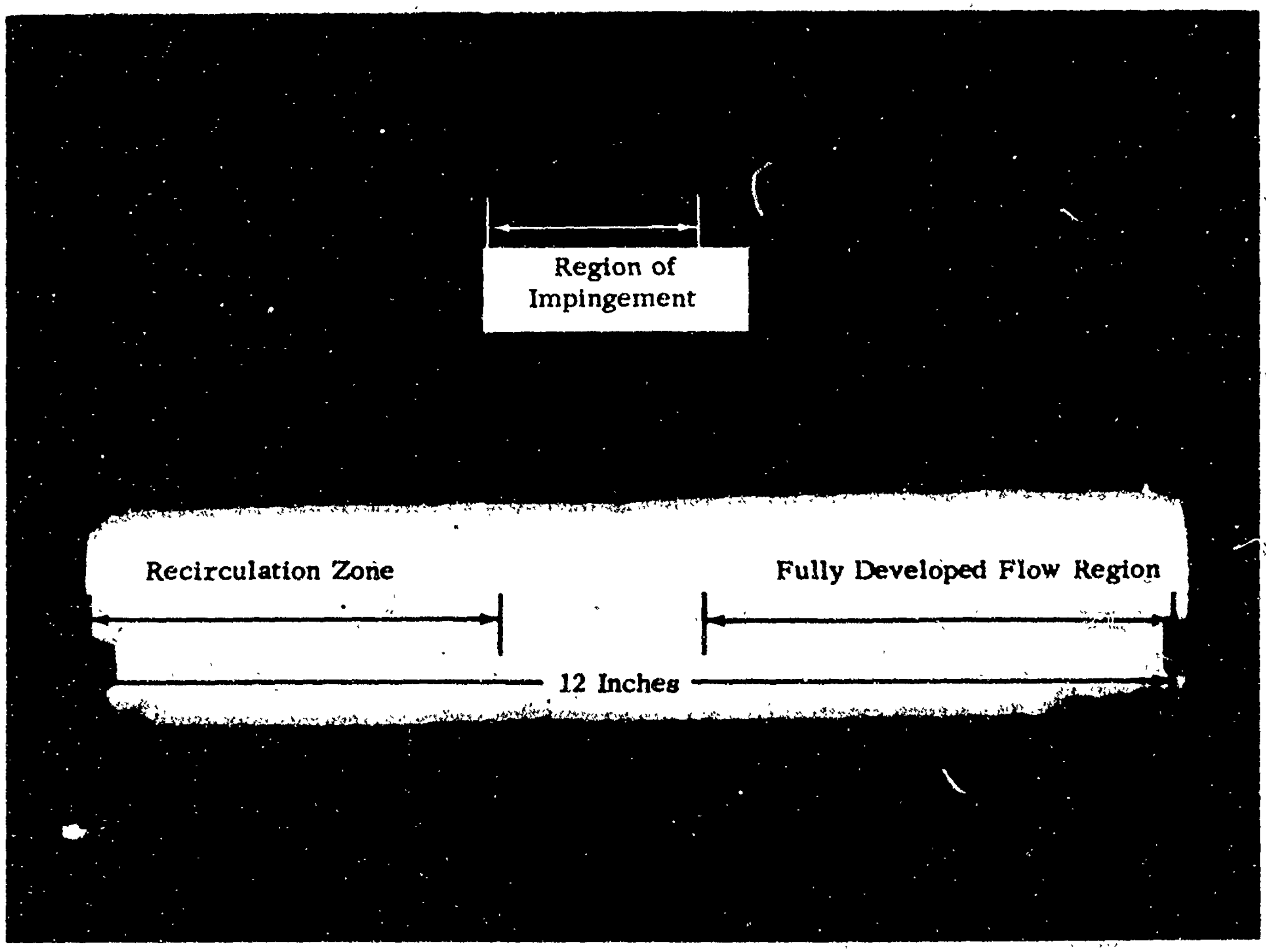


FIGURE 19 INJECTION NO. 1 - RECIRCULATION PATTERN REACTING FLOW

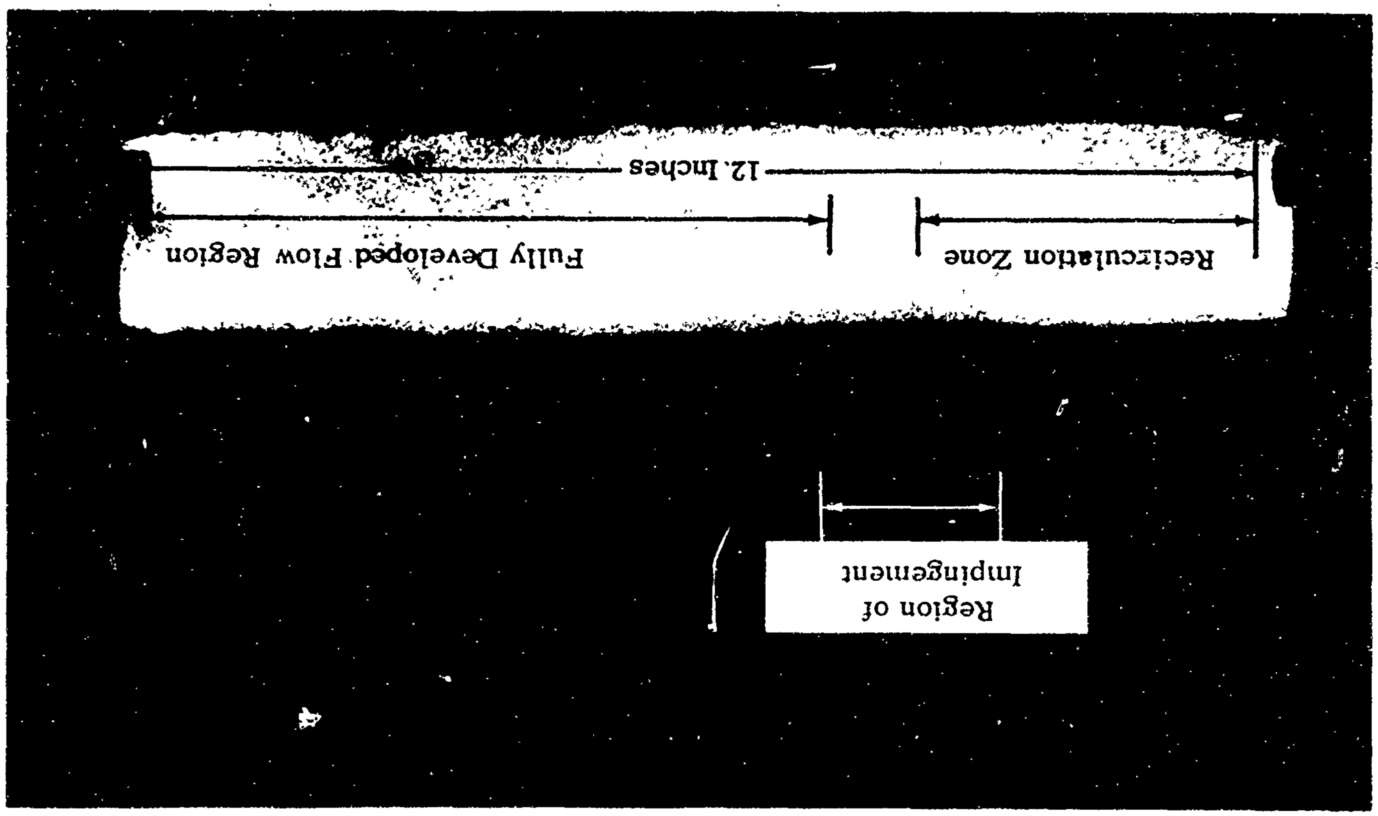


FIGURE 20 INJECTOR NO. 3 - RECIRCULATION PATTERN REACTING FLOW

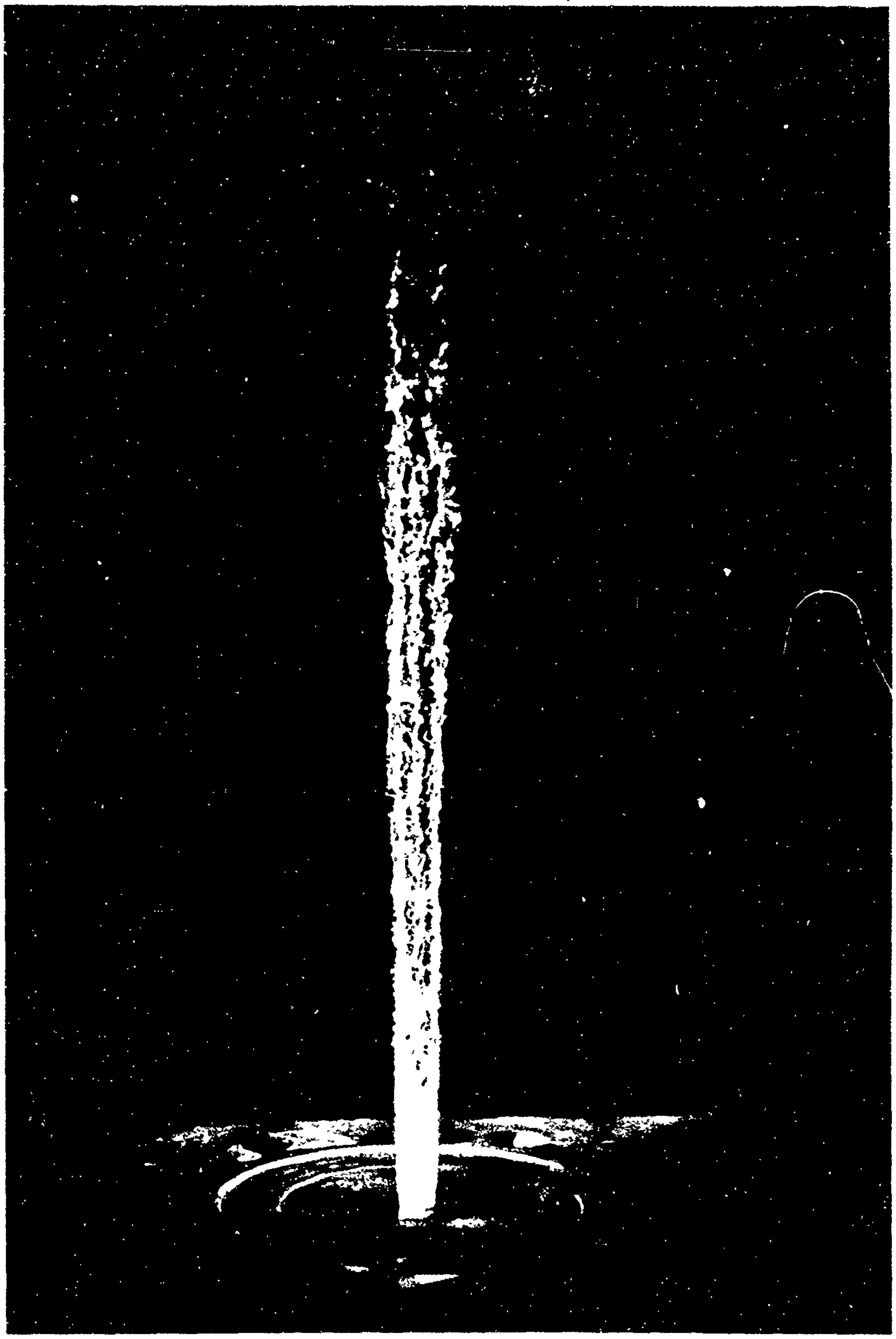


FIGURE 21 WATER FLOW THROUGH ANNULAR ORIFICE - NO NITROGEN CENTRALLY

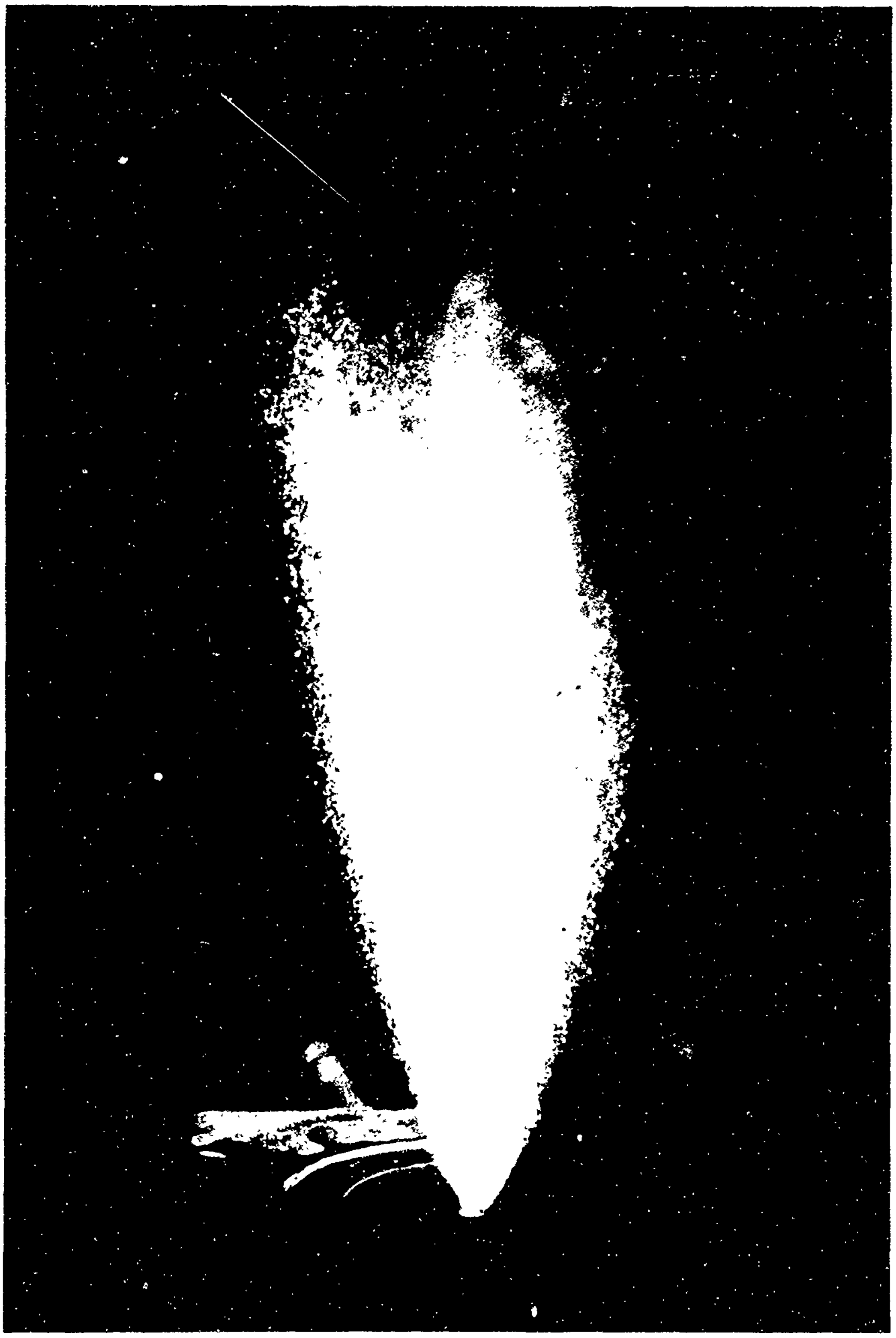


FIGURE 22 ANNULAR WATER JET - CENTRAL GASEOUS NITROGEN JET

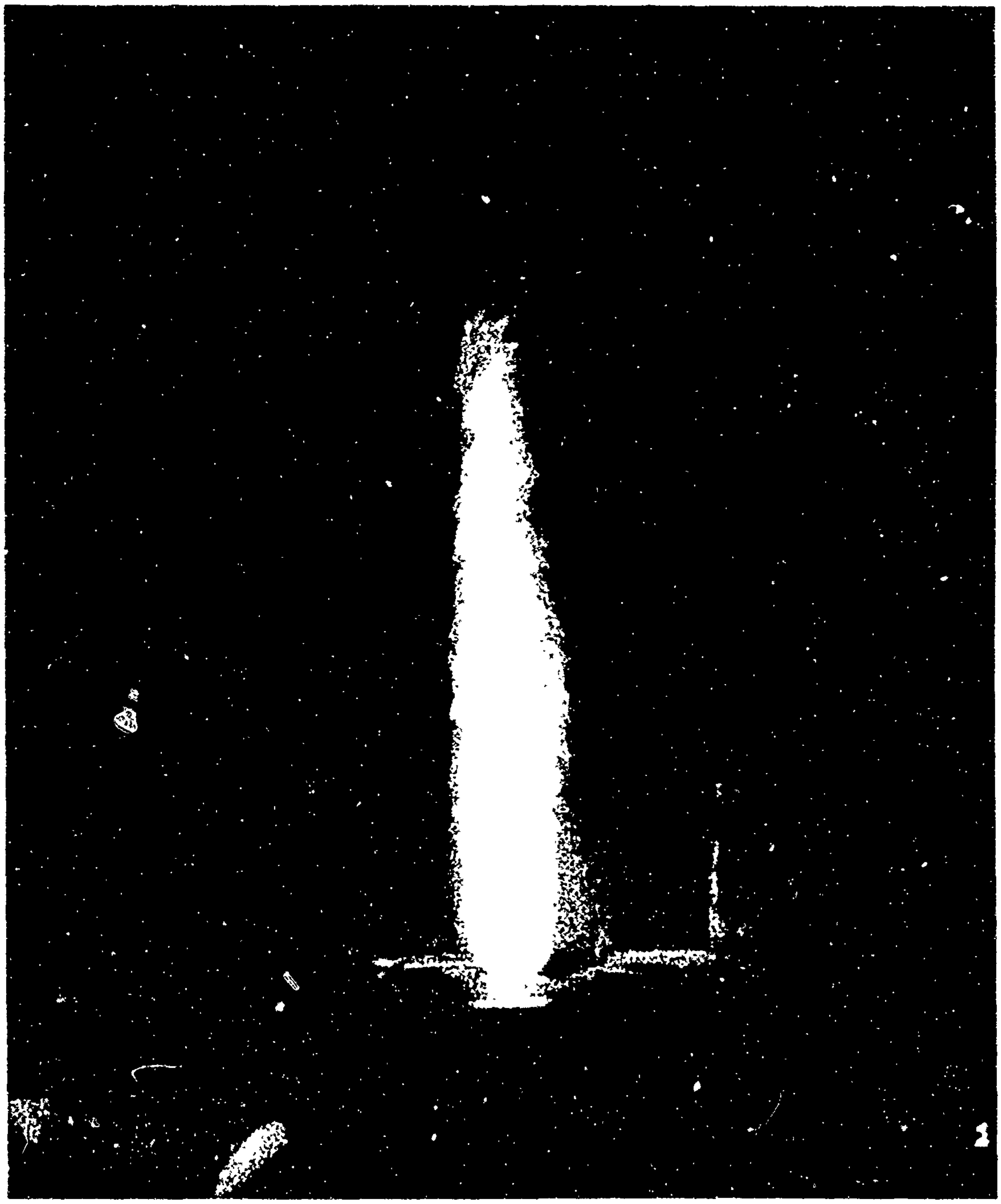


FIGURE 23 ANNULAR STREAM - LIQUID NITROGEN - 300 PSIA - CENTRAL STREAM - GASEOUS NITROGEN - 150 PSIA

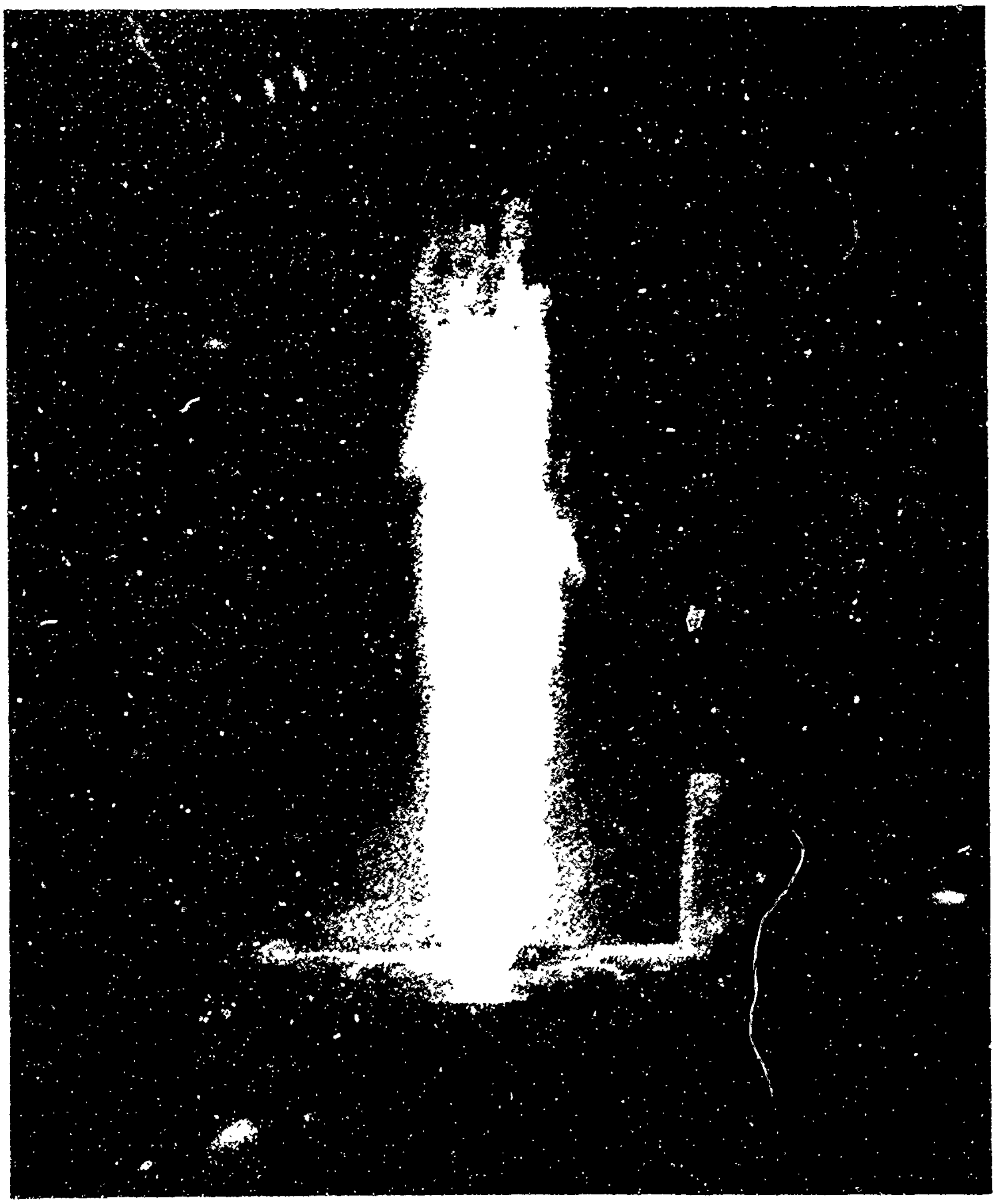


FIG. [illegible] ANNULAR STREAM - LIQUID NITROGEN, PRESSURE
[illegible] STREAM [illegible] PSIA

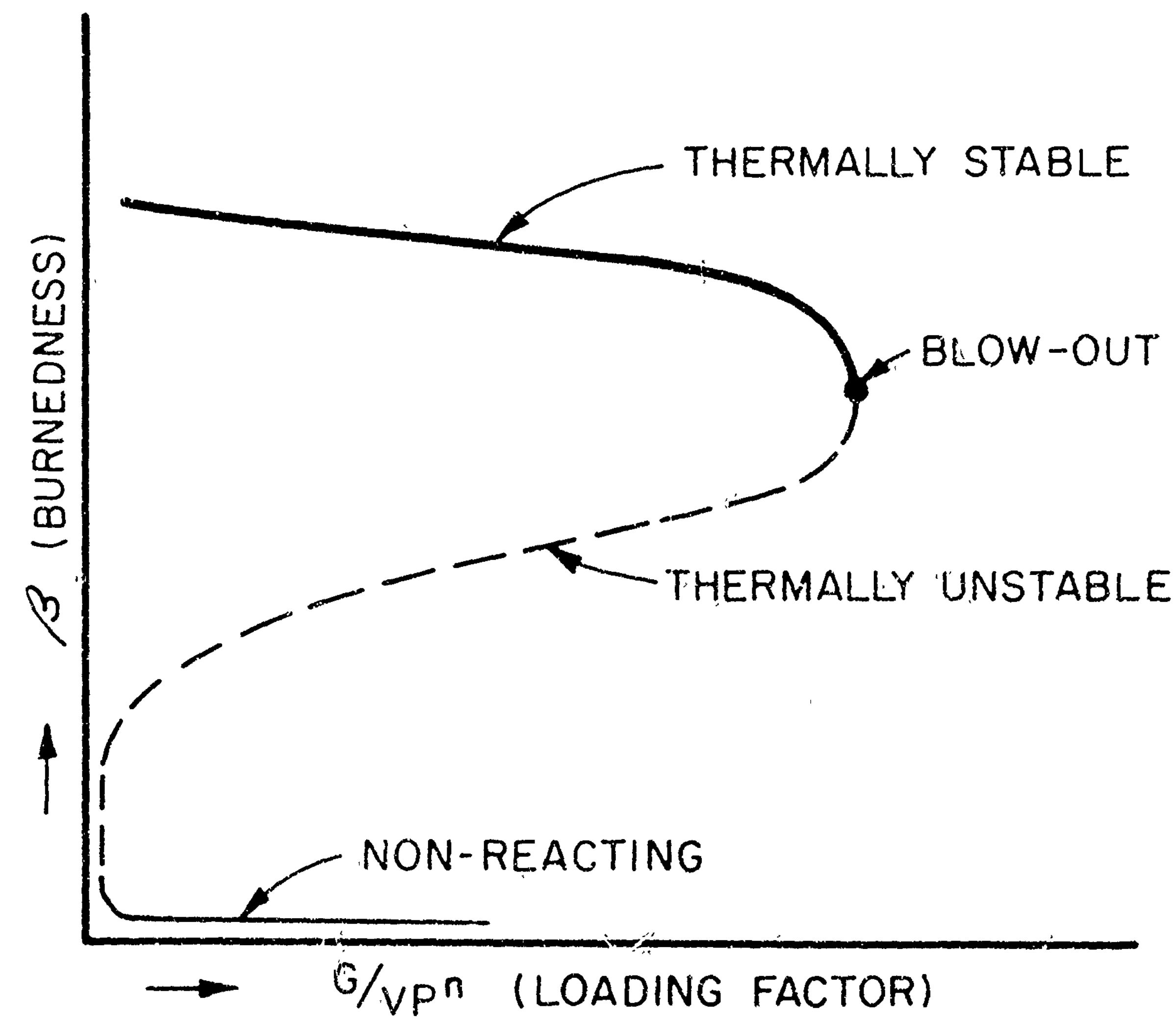


FIGURE 25 WELL-STIRRED REACTOR OPERATING CURVE

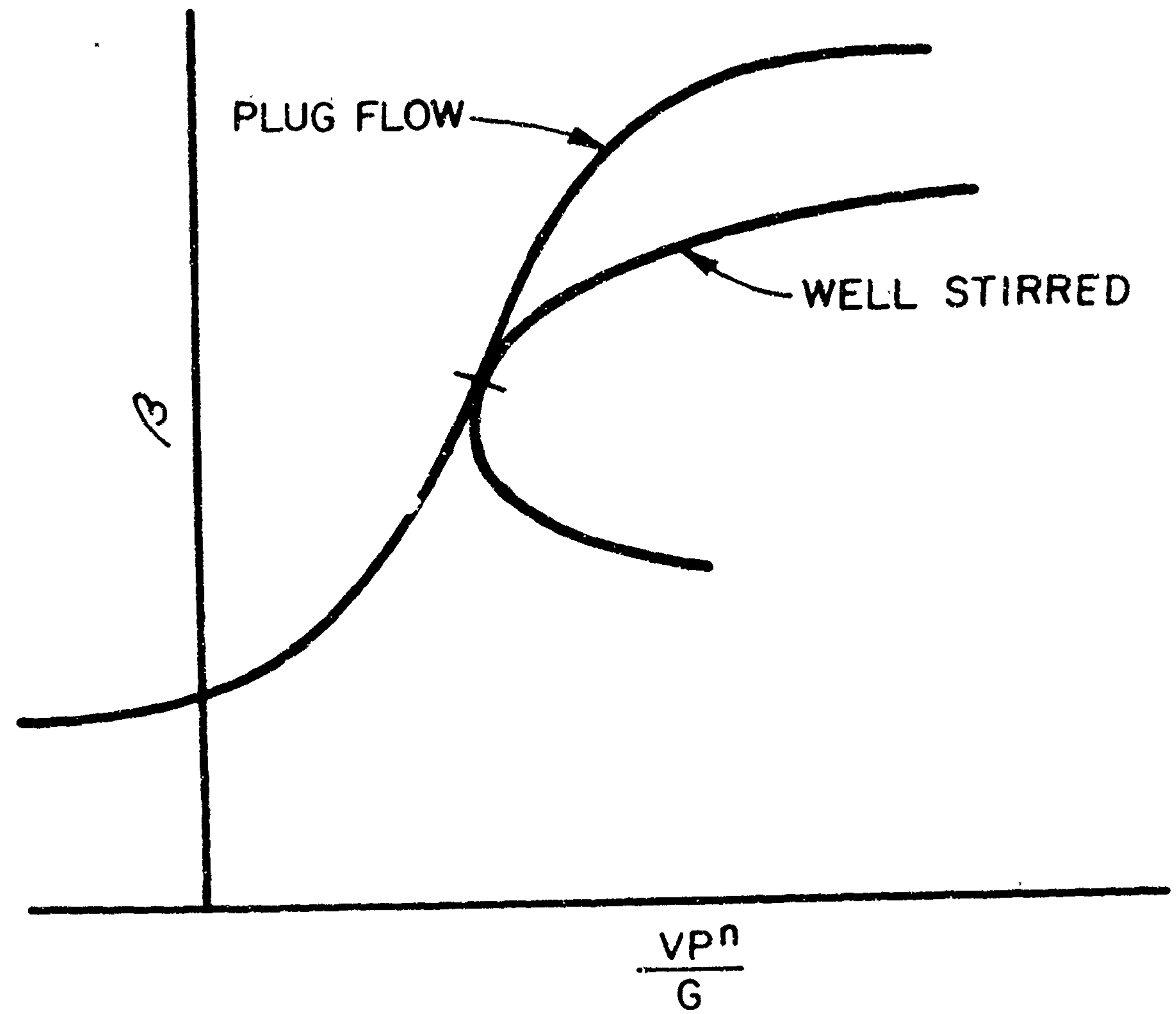


FIGURE 26 COMBINED PLUG-FLOW AND WELL-STIRRED REACTOR CURVES

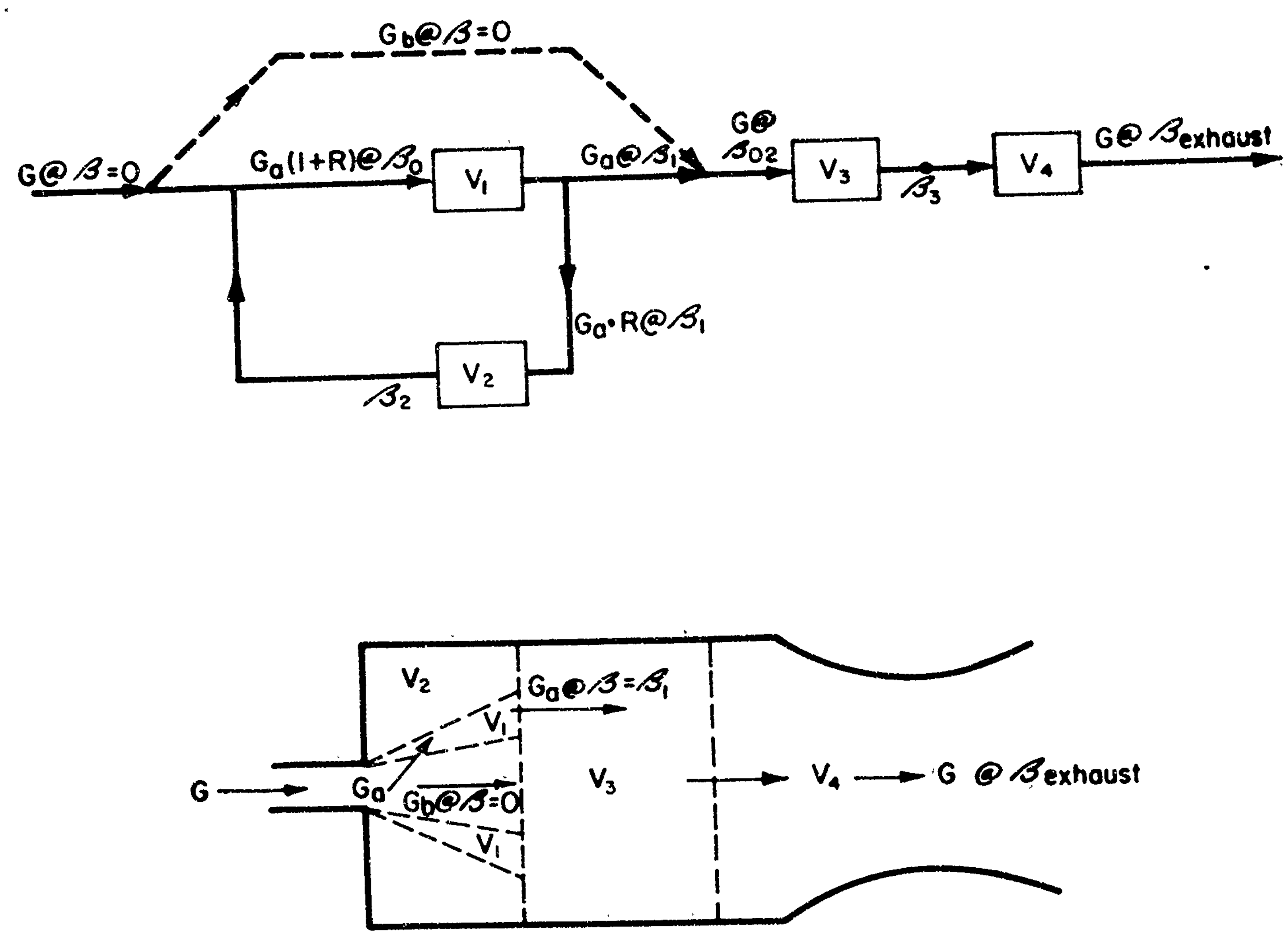


FIGURE 27 SCHEMATIC ANALOG OF ROCKET COMBUSTOR

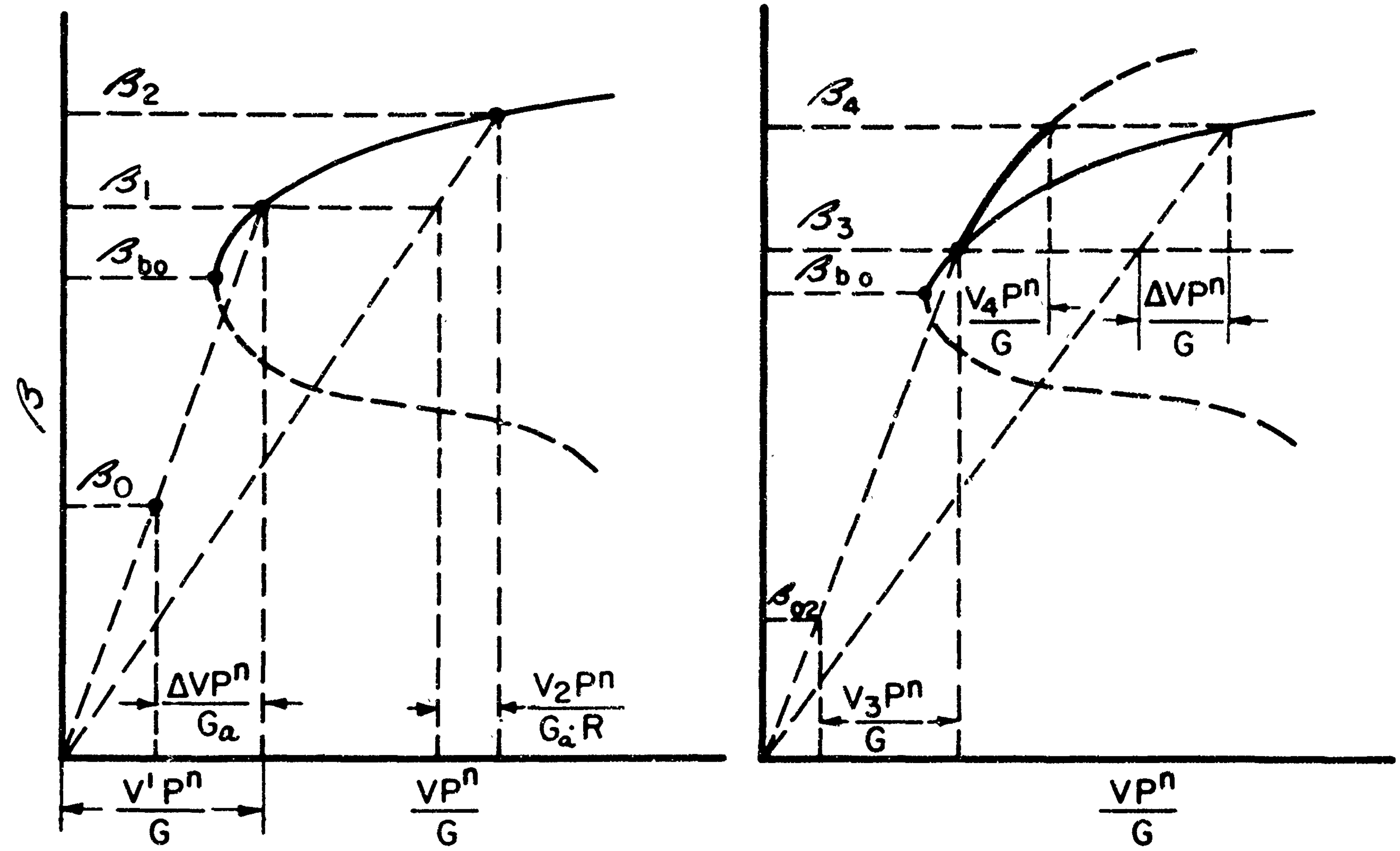


FIGURE 28 DEVELOPMENT OF ANALOG COMBUSTOR

UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1 ORIGINATING ACTIVITY (Corporate author) Arthur D. Little, Inc. Cambridge, Mass. 02140	2a REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b GROUP

3 REPORT TITLE

FLAME-PILOTING MECHANISMS IN LIQUID PROPELLANT ROCKET ENGINES

4 DESCRIPTIVE NOTES (Type of report and inclusive dates)

Scientific Interim

5 AUTHOR(S) (Last name, first name, initial)

Aubrey C. Tobey and E. Karl Bastress

6 REPORT DATE	7a TOTAL NO. OF PAGES	7b. NO. OF REFS
October 1966	74	16

8a CONTRACT OR GRANT NO. AF49(638)-1120	9a. ORIGINATOR'S REPORT NUMBER(S) 64601
b PROJECT NO 9711-01	
c 61445014	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)
d 681308	AFOSR 67-0938

10 AVAILABILITY/LIMITATION NOTICES

Distribution of this document is unlimited.

11 SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
TECH, OTHER	Dept. of the Air Force Office of Scientific Research (SREP)

13 ABSTRACT

This program was directed toward the systematic investigation of flame-piloting mechanisms in liquid rocket engines and their role in combustion instability. The mechanism which gained the most attention was recirculation. The program included experimental analysis of rocket engine performance as affected by external disturbances, minor geometry changes, and propellant state changes; and theoretical analysis of mixing in chemically reacting systems.

Principal conclusions drawn from the results of the program are:

a. Characteristics of the injector and of the propellants are principal factors in the chemical kinetics and fluid dynamics within rocket engine combustors.

b. For the rocket engine and perturbation technique used, engine stability is unaffected by perturbations of the recirculation zone. This does not necessarily eliminate the process of recirculation as a factor in engine stability

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
Flame-piloting						
Recirculation						
Combustion						
Combustion Instability						
Liquid Propellants						
Rocket Engines						

INSTRUCTIONS

1. ORIGINATING ACTIVITY: Enter the name and address of the contractor, subcontractor, grantee, Department of Defense activity or other organization (*corporate author*) issuing the report.

2a. REPORT SECURITY CLASSIFICATION: Enter the overall security classification of the report. Indicate whether "Restricted Data" is included. Marking is to be in accordance with appropriate security regulations.

2b. GROUP: Automatic downgrading is specified in DoD Directive 5200.10 and Armed Forces Industrial Manual. Enter the group number. Also, when applicable, show that optional markings have been used for Group 3 and Group 4 as authorized.

3. REPORT TITLE: Enter the complete report title in all capital letters. Titles in all cases should be unclassified. If a meaningful title cannot be selected without classification, show title classification in all capitals in parenthesis immediately following the title.

4. DESCRIPTIVE NOTES: If appropriate, enter the type of report, e.g., interim, progress, summary, annual, or final. Give the inclusive dates when a specific reporting period is covered.

5. AUTHOR(S): Enter the name(s) of author(s) as shown on or in the report. Enter last name, first name, middle initial. If military, show rank and branch of service. The name of the principal author is an absolute minimum requirement.

6. REPORT DATE: Enter the date of the report as day, month, year; or month, year. If more than one date appears on the report, use date of publication.

7a. TOTAL NUMBER OF PAGES: The total page count should follow normal pagination procedures, i.e., enter the number of pages containing information.

7b. NUMBER OF REFERENCES: Enter the total number of references cited in the report.

8a. CONTRACT OR GRANT NUMBER: If appropriate, enter the applicable number of the contract or grant under which the report was written.

8b, 8c, & 8d. PROJECT NUMBER: Enter the appropriate military department identification, such as project number, subproject number, system numbers, task number, etc.

9a. ORIGINATOR'S REPORT NUMBER(S): Enter the official report number by which the document will be identified and controlled by the originating activity. This number must be unique to this report.

9b. OTHER REPORT NUMBER(S): If the report has been assigned any other report numbers (*either by the originator or by the sponsor*), also enter this number(s).

10. AVAILABILITY/LIMITATION NOTICES: Enter any limitations on further dissemination of the report, other than those imposed by security classification, using standard statements such as:

(1) "Qualified requesters may obtain copies of this report from DDC."

(2) "Foreign announcement and dissemination of this report by DDC is not authorized."

(3) "U. S. Government agencies may obtain copies of this report directly from DDC. Other qualified DDC users shall request through ________________."

(4) "U. S. military agencies may obtain copies of this report directly from DDC. Other qualified users shall request through ________________."

(5) "All distribution of this report is controlled. Qualified DDC users shall request through ________________."

If the report has been furnished to the Office of Technical Services, Department of Commerce, for sale to the public, indicate this fact and enter the price, if known.

11. SUPPLEMENTARY NOTES: Use for additional explanatory notes.

12. SPONSORING MILITARY ACTIVITY: Enter the name of the departmental project office or laboratory sponsoring (*paying for*) the research and development. Include address.

13. ABSTRACT: Enter an abstract giving a brief and factual summary of the document indicative of the report, even though it may also appear elsewhere in the body of the technical report. If additional space is required, a continuation sheet shall be attached.

It is highly desirable that the abstract of classified reports be unclassified. Each paragraph of the abstract shall end with an indication of the military security classification of the information in the paragraph, represented as *(TS)*, *(S)*, *(C)*, or *(U)*.

There is no limitation on the length of the abstract. However, the suggested length is from 150 to 225 words.

14. KEY WORDS: Key words are technically meaningful terms or short phrases that characterize a report and may be used as index entries for cataloging the report. Key words must be selected so that no security classification is required. Identifiers, such as equipment model designation, trade name, military project code name, geographic location, may be used as key words but will be followed by an indication of technical context. The assignment of links, rules, and weights is optional.

UNCLASSIFIED
Security Classification